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APPENDIX F

AIRCRAFT/MISSILE SEPARATION DYNAMICS

1. Discussion of Problem

One of the central questions in establishing feasibility of the proposed air launch mission concerns the dynamics associated with reliable separation of the pod containing the Minuteman booster from the B-58A aircraft. The preliminary analysis of the air launch and separation discussed below shows that these are feasible, though considerable development work will be necessary to confirm details of the approach chosen.

The launching sequence is as follows: a mild pull-up maneuver is followed by release of the loaded pod, with application of an impulsive moment to impart a negative pitch rate to the loaded pod. During free fall, separation of the pod halves occurs followed by ignition of the Minuteman. It is simpler, for an engineering evaluation, to analyze the launch sequence in reverse chronological order, starting with the desired booster attitude at ignition and working back, to obtain the impulsive moment requirements and the maneuvering requirements for the aircraft. The booster velocity and altitude at ignition which are of prime importance in analyzing performance, have only a second order influence on the dynamic phenomena.

To predict the impulsive control moment requirements and their dependence on launching conditions, an elaborate digital calculation of possible launch trajectories has been carried out. This analysis is based on the following concepts:

a) At ignition, the angle of attack of the booster must be within prescribed limits in order that the destabilizing aerodynamic moments do

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not exceed the maximum control moments obtained from gimbaling of the rocket nozzles. At the nominal launch dynamic pressure (approximately 500 lb/ft^2), the maximum allowable angle of attack limits are, at worst, ±12 degrees. For the separation analysis, a much tighter requirement of ±4 degrees was chosen. This requirement was chosen to increase the apparent dependence of control moment on launch conditions, in order to define more sharply regions of best launch conditions. Thus, in the final results, the actual limitations of ±12 degrees appear as quite wide tolerances in possible launch conditions about the best launch conditions.

b) Separation of the pod from the Minuteman is achieved by using explosive bolts plus separation rockets to eject the sides of the pod away from the Minuteman. A pitch rate is imparted to the separate pod panels by delaying the firing signal to the aft bolts. The forward bolts fire first and the panels momentarily rotate about an aft-mounted hinge. The pod panels rapidly build up angle of attack so that large aerodynamic forces are created which aid in obtaining a clean separation.

c) During free fall of the pod prior to separation from the Minuteman, the pod is acted on by the complex flow fields beneath the launching aircraft. The incremental lift, drag, and moment coefficients arising from the interference flow field were estimated from wind tunnel tests of a similar, though not identical, aircraft-pod configuration. The dimensions shown in the table have been adjusted to full scale for the B-58A/pod combination. For the tabulated coordinates, x is the distance in feet measured from the fuselage nose parallel to the fuselage center line and z is the lateral distance in feet from the fuselage center line

The pod configuration, which is unstable in a uniform flow field, is further destabilized due to the interference flow field effects. However because of the large pod pitching moment of inertia and the low dynamic pressure, the angular accelerations arising from the destabilizing aerodynamic moments are small.

d) At separation of the pod from the launching aircraft, the pod has an initial pitch up rate and a positive angle of attack of approximately 6 degrees. The positive pitch up rate combined with the destabilizing moment tends to increase the angle of attack during free fall so that, if uncorrected, the Minuteman angle of attack at ignition is near or exceeds the ± 12 degree limits. For this reason an impulsive moment must be applied to the pod which is sufficient to overcome the initial pitch rate plus the pitch up tendency of the aerodynamic moments in order to rotate the pod downward from its initial launch angle of attack to the desired ± 4 degree orientation at ignition. It is possible to impart an initial pitch down rate if the airplane flies a complicated prelaunch maneuver. Maneuvers of this type were investigated and found to be unsuitable.

e) The prelaunch maneuver is selected on the basis of several criteria. First, it must not require violent control action or extreme coordination on the part of the pilot. Second, the launching conditions (primarily pitch rate) should be selected so that the required impulsive moment is not excessively large. Third, there should be wide tolerances about the nominal launch attitude to account for pilot errors and possible variations in the applied control moment. Fourth, the attitude and

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angular velocity of the aircraft during the early stages of separation should be selected so that the separation rate is maximized. Since the loaded pod weighs approximately 75,000 pounds and the aircraft between 80,000 and 100,000 pounds, at separation the lateral acceleration of the aircraft jumps by a large amount before the re-establishment of trim, giving an unusually large separation acceleration.

2. Design Approach

The impulsive control moment can be applied to the pod in several ways:

a) Oversize stabilizing fins can be mounted on the pod to give a large restoring moment to reduce the angle of attack at ignition. This method of approach is unsatisfactory because even at low altitude where the stabilizing fins are most effective, the required area and resultant weight are unreasonable. Furthermore, the pod would interact very strongly with the nonlinear interference flow field resulting in adverse coupling with the aircraft attitude.

b) Small rocket engines placed some distance from the center of gravity of the pod can be utilized to provide an impulsive moment. Although practical, this technique results in a heavier, more complex, and less reliable system than the one chosen.

c) A "springy" hook attachment between aircraft and pod (in addition to the rigid supports for carrying the pod) which gives a force proportional to velocity and distance between attach points and which breaks at a specified separation distance, has been selected to provide the desired impulsive pitch-down moment. This technique was chosen as the simplest and lightest that adequately met requirements. Since it depends more on passive parts than other systems, it has more reliability. Extensive analysis was made of this system.

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Over 200 launch trajectories were analyzed for the effects of the magnitude and duration of hook force application and initial pod altitude, velocity, lateral acceleration, and pitch attitude. These calculations were based on a simultaneous solution of four second-order, nonlinear, total differential equations which describe the two-dimensional motion of the aircraft and pod. Because of difficulties in programming, the interference flow field effects were not included in the machine computations. However, these incremental forces and moments were computed by hand for several trajectories and their contributions to motion of the pod were estimated. The interference flow field had no effect on those trajectories that rapidly diverged to large positive or negative angles of attack. For those trajectories that tended to dwell near zero degrees angle of attack, these terms were important; but their effect could be compensated for by a small change in hook force or initial angle of attack. Thus, the nature of the trajectories was essentially unaffected by the presence of the interference flow field, but the choice of hook force or initial angle of attack required to obtain the desired behavior was altered slightly.

From these studies it was apparent that the pod should be launched at the highest possible altitude. Data provided by Convair for a 2.5 g pull up of a B-58A carrying a store similar to the pod indicates a maximum altitude of 60,000 feet at Mach 2.0 is obtainable. For these conditions the dependence of separation distance, separation time, and hook force on the aircraft/pod combination lateral acceleration at launch is summarized in the following table.

Lateral Acceleration (g)	Separation Distance (ft)	Separation Time (sec)	Hook Force (lb)
0.5	25	2.0	10,000
1.0	40	1.7	20,000
1.5	55	1.5	30,000
2.0	80	1.3	45,000

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The separation distance tabulated is the average distance from the aircraft at which the pod angle of attack is within the ± 4 degree limits. The dwell time (i.e., duration of time for which the pod is within ± 4 degrees angle of attack) varied between 0.2 and 2.0 seconds. All of the tabulated quantities represent nominal values. There is a considerable variation about the nominal (as much as 100 percent) primarily dependent on the initial angle of attack. Figures F-1 to F-4 illustrate typical trajectories and their dependence on initial conditions and hook force. In each figure the pod angle of attack has been plotted as a function of separation distance. The tic marks delineate successive time intervals of 0.1 second. From the results of this analysis a launch condition of 60,000 feet altitude, Mach 2, and a lateral acceleration of 1 g appears to be the most satisfactory condition.

The analyses did not include the complete dynamic response of the aircraft since there was insufficient information available regarding the damping and control derivative. This deficiency was overcome by computing the trajectories for each set of initial conditions twice. In the first calculation the aircraft was assumed to pitch upwards after launch at a constant rate equal to that existing prior to launch. In the second calculation the aircraft pitch rate was computed by assuming that the aerodynamic moments acting on the aircraft were constant before and after launch. This assumption neglects the aerodynamic damping and control deflection to re-establish trim. These two assumptions provide conservative limits and the actual trajectory should be within these extremes. A rigid aircraft and loaded pod structure was assumed throughout.

It is difficult to assess the reliability of the air launch technique primarily because of the lack of quantitative data for the effects of:

a) Interference flow field

b) Aircraft dynamics

c) Nonrigid structure

d) Allowable structure load.

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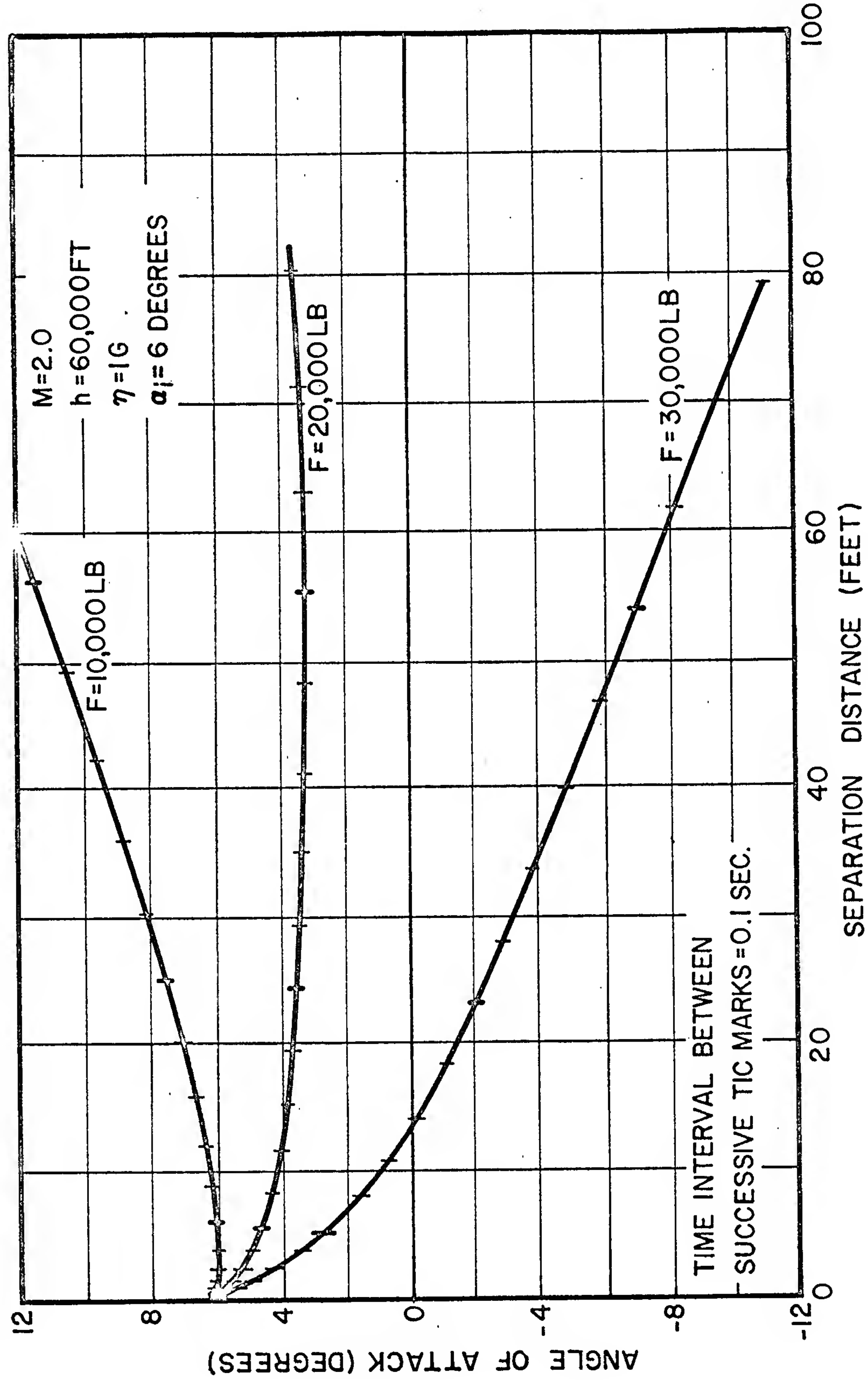

Figure F-1. Angle of Attack Versus Separation Distance, Effect of Hook Force

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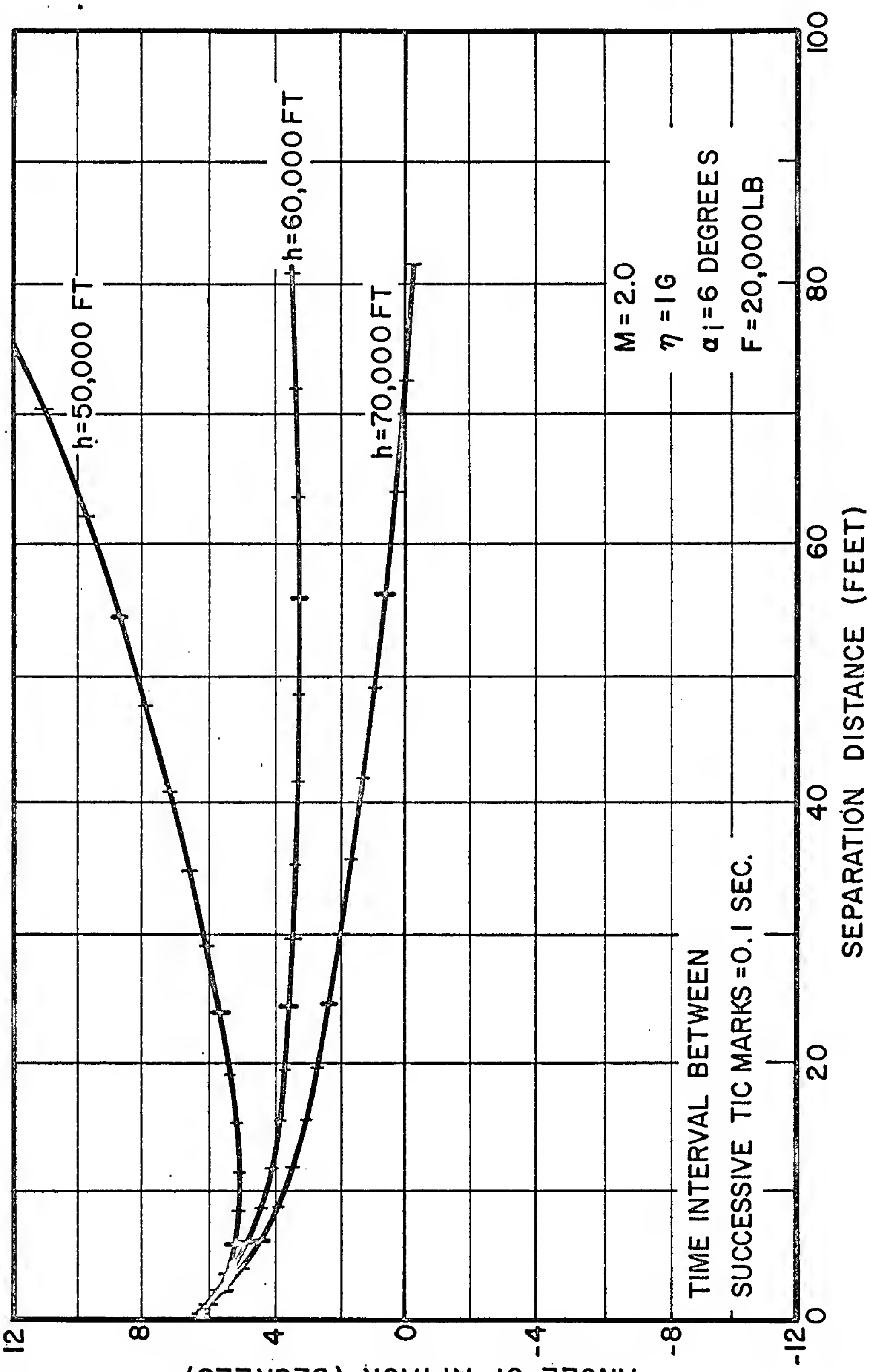

Figure F-2. Angle of Attack Versus Separation Distance, Effect of Altitude

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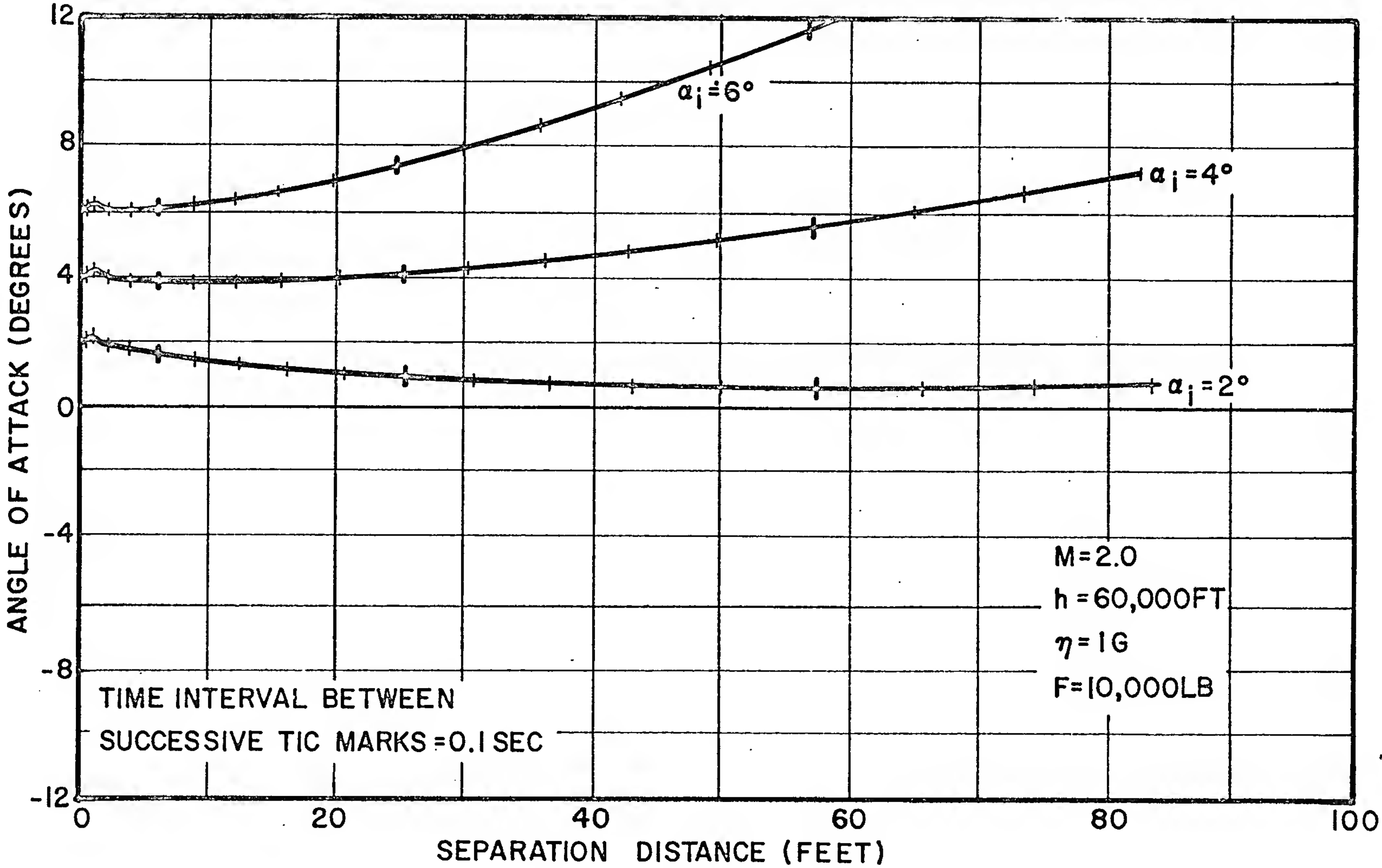

Figure F-3. Angle of Attack Versus Separation Distance, Effect of Initial Angle of Attack

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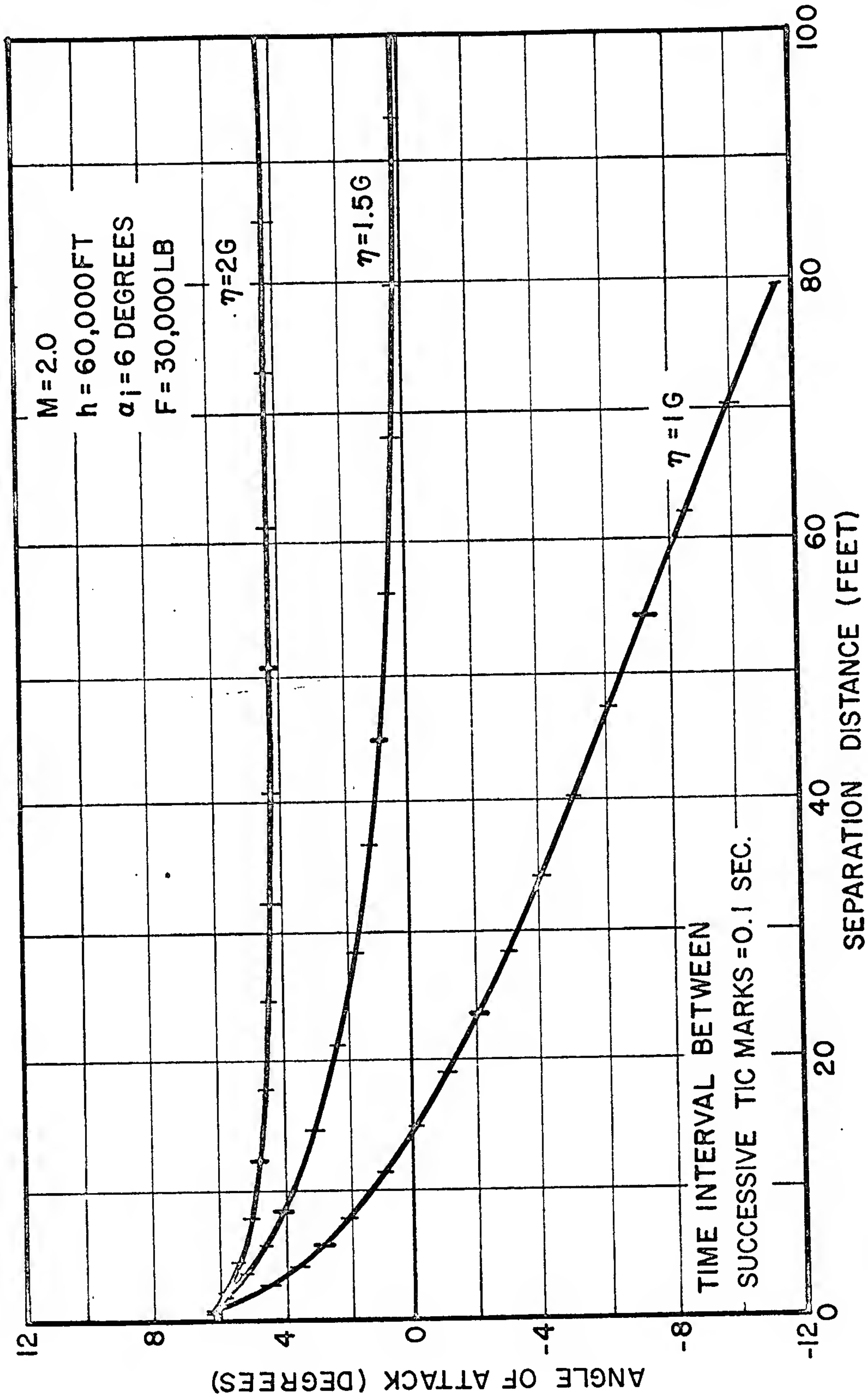

Figure F-4. Angle of Attack Versus Separation Distance, Effect of Lateral Acceleration

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In each case (possibly with the exception of c) above) conservative assumptions were used to circumvent the lack of information. Even with these conservative assumptions, extremely wide tolerance in allowable launch conditions and hook forces were obtained. Thus, the approach is clearly feasible, although extensive wind tunnel tests, analog simulation of the entire nonrigid aircraft/pod configuration, and probably some flight testing, will be required for detailed confirmation of the method.

Time did not permit remaking these separation calculations for the best launch conditions as determined by the final estimates of B-58A performance and the best flight path inclination for MM. These "best trajectory" values are $M = 1.6$ at 56,000 feet altitude and velocity inclination of 20° to the horizontal. However, it is quite evident that a satisfactory separation condition is possible for this best trajectory.

No analysis was made of pod yawing motion during the separation process. Compensation for initial yaw angle and rate at release can easily be provided if necessary. Finally, we can assume that no difficulty will arise from initial rolling action at release.

The pilot's problem is not simple. He must attain a preset altitude, flight path angle, pitch rate (lateral acceleration), and angle of attack in a pullup maneuver taking approximately 40 seconds from the level flight starting point. However, the deviation allowable on each parameter is so large that we believe the maneuver to be quite feasible.

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APPENDIX G

TRAJECTORY STUDIES

The objective of the trajectory studies was to determine the performance of the air-launched Minuteman (MM) in terms of payload weight orbited for the operational system.

Because the aircraft may not be able to obtain desirably large upward flight path angles at extreme altitudes, a pull-up maneuver will generally be necessary just prior to pod release. Hence the determination of the performance of an air-launched satellite booster system must account for the fact that the launch path angle, velocity, and altitude are all variable to some extent. This study makes a parametric evaluation of these variables in order to optimize the system performance.

The launching of the reconnaissance vehicle from its supersonic carrier produces many other problems. Among these is the fact that the externally stored pod is in a very complex flow field around the airplane. It was therefore necessary to initiate flow field studies to determine the proper arrangement of pod and airplane. It was also necessary to study carefully the effect of the flow field on the stability, control and separation dynamics of the missile and opened pod. These studies are also reported herein.

1. Summary

The trajectory studies use the Wing II Minuteman performance and weight data Initial conditions at Minuteman ignition correspond to the B-58A airplane in a 2g pull-up from Mach 2.4 level flight at 45,000 foot altitude. The results are in terms of payload weights in polar orbit for 1, 0.75, 0.50, 0.25 and 0 percent

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elliptical orbits injected 95 degrees of earth angle downrange from a 65 nautical mile altitude perigee. Payload weights include everything forward of the Minuteman third stage engine, i.e., guidance, transition section and satellite. The results are:

a) Trajectory design 3 currently gives the highest payload weights in orbit of all trajectories designed. Trajectory design 3 holds the missile attitude constant over first stage burn and operates with constant pitch rates for second and third stage burn. The performance is very sensitive to the missile pitch angle with respect to the vertical at ignition (see Figure G-1). See Figure G-27 for definition of terms.

b) Payload weights are very sensitive to injected altitude.

Ellipse (percent)	Injection Altitude (n mi)	Nominal Payload Weight (lb)
0 circular	65	1518
0.25	75	1436
0.50	85	1346
0.75	95	1251
1.0	105	1150

These payloads result from computer runs at near optimum trajectory conditions (see Figures G-2 and G-9 through G-23). Orbits are for a 65 nautical mile perigee altitude with injection at 95° from perigee and with a 90° inclination (polar).

c) For an inclined orbit (70°) earth rotation effect, launching at 35° latitude gives the following payload increments for a 0.5 percent elliptical orbit (see Figure G-3).

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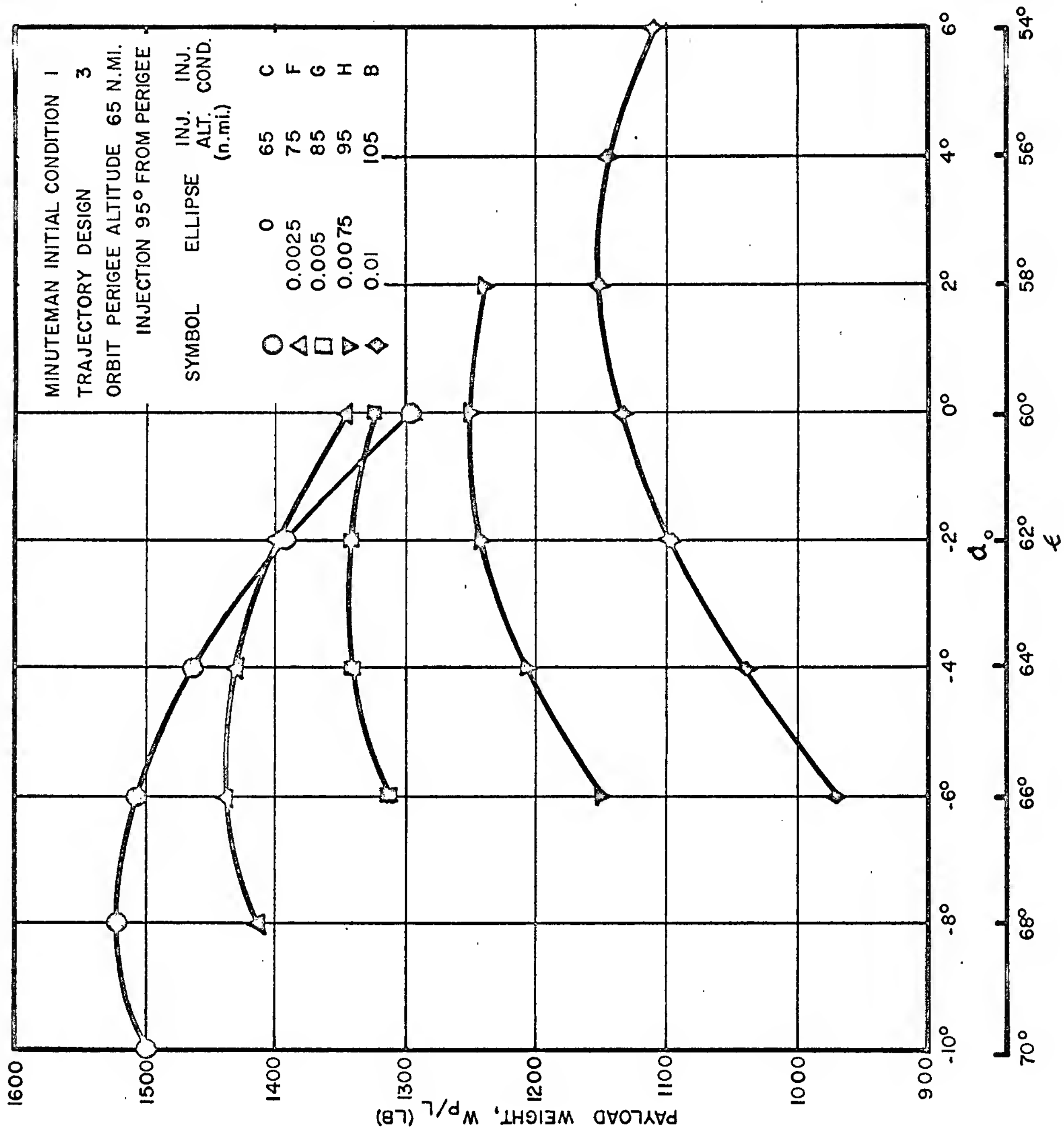

Figure G-1. Air Launch Minuteman Performance Injecting Into Low Altitude Orbits

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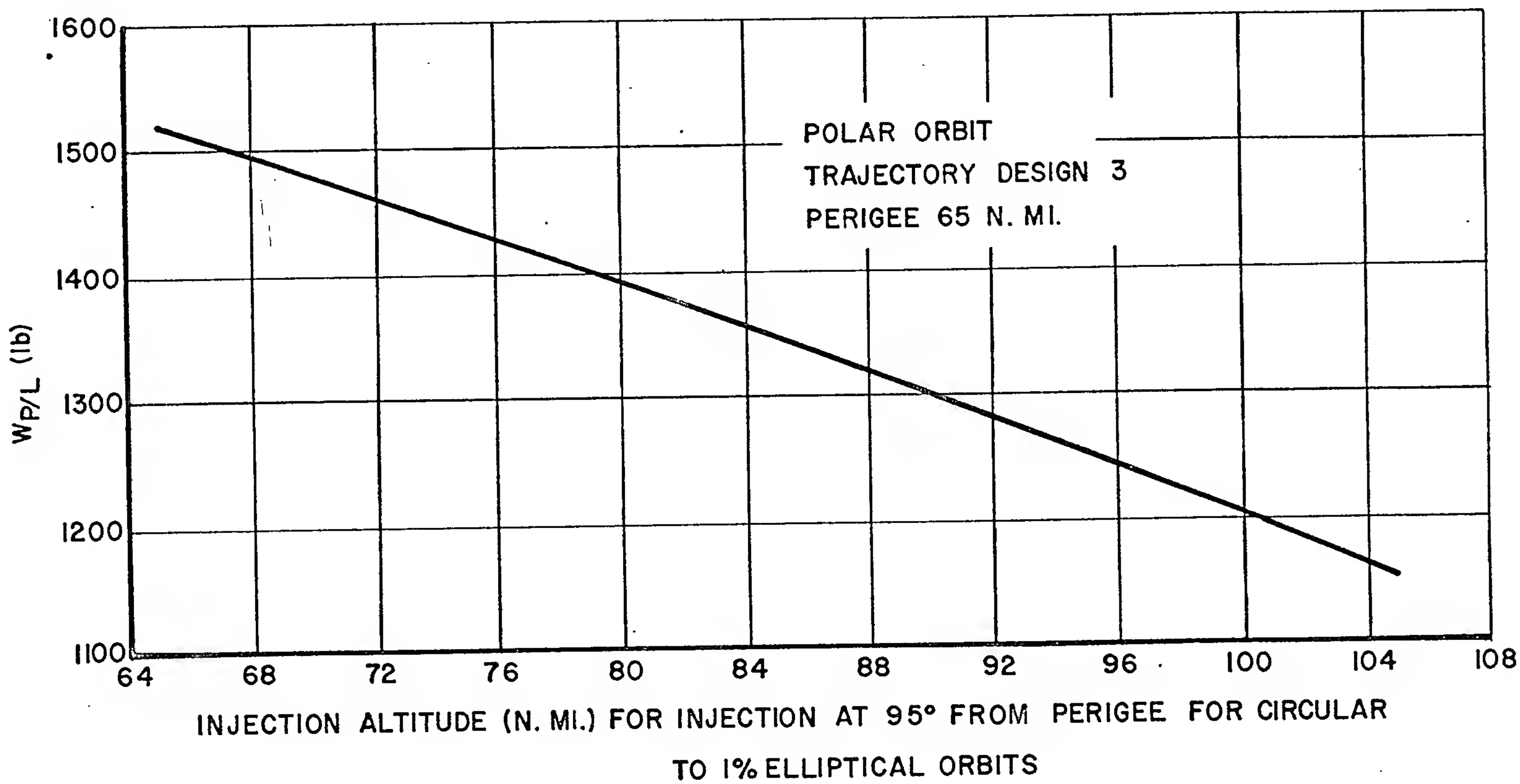

Figure G-2. Air Launch Minuteman Performance Variation with Orbital Injection Altitude

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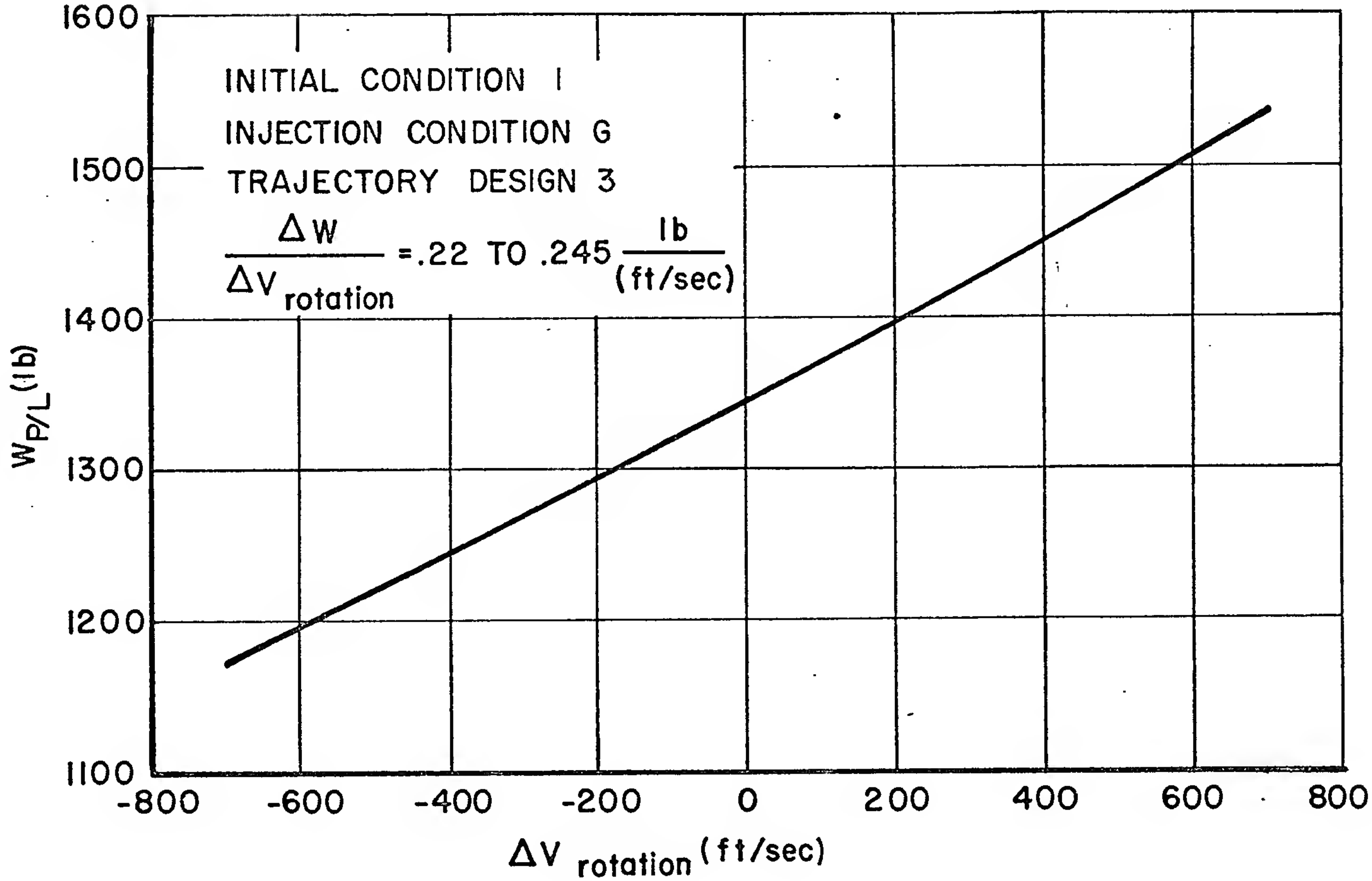

Figure G-3. Effect of Earth Rotation Velocity ($\Delta V_{rotation}$) on Air Launched Minuteman Performance

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Launch	$\Delta V_{rotation}$ (ft/sec)	Payload Weight Increment (lb)
Southeast	+700	+190
Southwest	-700	-173

d) Launching the Minuteman from a 2 g pull-up from level Mach 2.0 (instead of 2.4) flight decreases payload capability.

Ellipse (percent)	Payload Increment (lb)
0.25	-146
0.50	-129

e) Results for trajectory simulation with a coast period between Minuteman second stage burnout and third stage ignition indicate that substantia payload weight increases are possible. One simulation increased payload weight 240 pounds over the best comparable continuous burn trajectory.

For the Minuteman weapon system, the 3-sigma performance variation is 200 ft/sec at burnout. This corresponds to a 51-pound decrease in payload capability for these low orbits.

A typical trajectory (1 percent orbital ellipticity) was tried for horizonta launch (no pull up) at Mach 2.40. The payload in this case was only 728 pounds (see Figures G-24 through G-26).

2. Minuteman Trajectories, Two-Dimensional Simulation Studies

a. Initial Conditions

Air-launched Minuteman trajectories are defined to begin with first stage ignition. This ignition takes place 0.5 to 1.5 seconds after release from

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the B-58A. During this short time, changes take place in dynamics, position, velocity and flight path angle which fortunately have small effect on performance as shown in Table G-1. Therefore the principal initial parameters affecting Minuteman performance are determined by the conditions at the instant of release from the B-58A. These are therefore examined in detail.

b. Azimuth

The B-58A airplane capability with the Minuteman store attached is best used by flying the airplane along a heading (azimuth) in the plane of the desired orbit (see Figure G-4). Errors in airplane flight path heading at Minuteman release are expected to be less than 0.5 degree with the B-58A flying on autopilot. Degradation of performance due to this misalignment of the velocity vector (δ_{az}) will be extremely small since

1) The degrading effect on the horizontal component of the velocity vector (Δv_h) in the plane of the orbit is

$$\Delta v_h = v_{h_{in}} (1 - \cos \delta_{az})$$

where $v_{h_{in}}$ = initial horizontal velocity, δ_{az} = misalignment of the velocity vector in azimuth and it is obvious that for $\delta_{az} < 0.5$ degree, $\cos \delta_{az}$ is very close to 1.0

2) The vertical component of the velocity vector is not affected

3) The corrective steering and thrusting required to compensate for the lateral velocity (maximum 12 ft/sec) is small and well within normal steering and thrust losses for Minuteman flying through atmospheric winds.

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Table G-1. Typical Variations in Initial Conditions for Minuteman Trajectory After Release from B-58A. (M = 2.0, altitude = 60,000 ft)

Airplane Release Conditions	Initial Condition Parameters	Time (sec)* 0	0.5	1.0	1.5
1.5 g pull-up	β (deg)	70.0	70.3	70.7	71.1
α_i = 8 degrees	h (ft)	60,000	60,488	60,985	61,492
hook force = 30,000 lb	v_a (ft/sec)	1936	1928	1919	1910
	Separation distance (ft)	0	8.8	36.9	84.6
1.0 g pull-up	β (deg)	70.0	70.4	70.8	71.2
α_i = 6.0 degrees	h (ft)	60,000	60,486	60,974	61,467
hook force = 20,000 lb	v_a (ft/sec)	1936	1928	1919	1910
	Separation distance (ft)	0	5.8	24.4	56.0
0.5 g pull-up	β (deg)	70.0	70.4	70.8	71.2
α_i = 4.0 degrees	h (ft)	60,000	60,483	60,963	61,442
hook force = 10,000 lb	v_a (ft/sec)	1936	1928	1919	1911
	Separation distance (ft)	0	2.8	11.9	27.4

*Time from separation from B-58A to Minuteman ignition.

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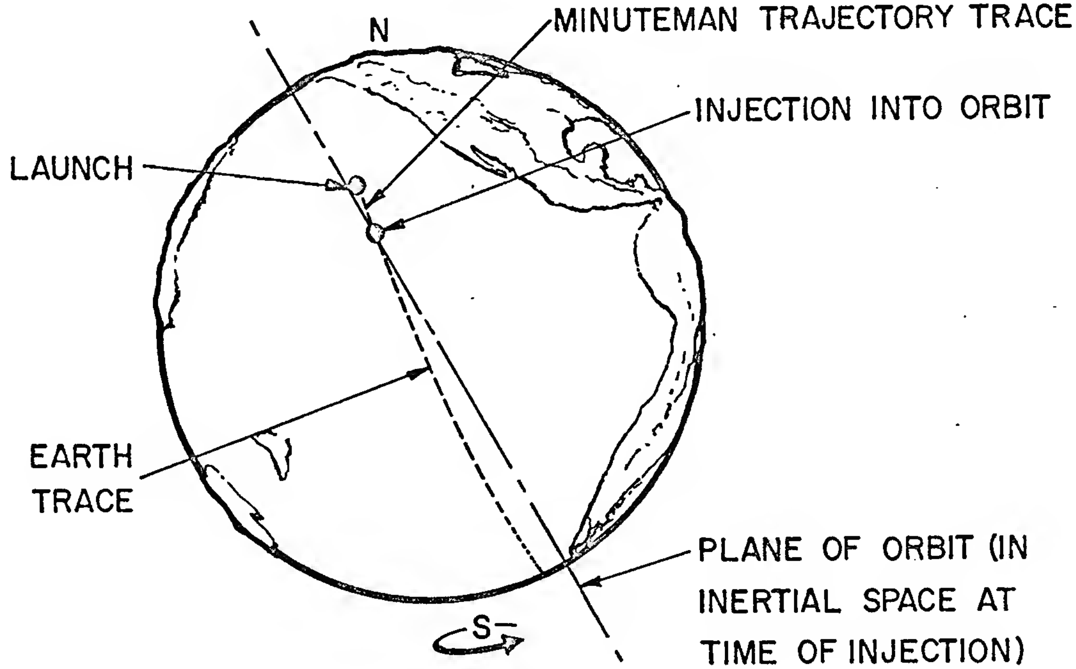

Figure G-4. Typical Flight Path Heading (Azimuth) Diagram for Airplane During Launch Operation

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c. Altitude, Velocity and Angle

The B-58A is flown at maximum altitude and Mach number for straight and level flight with the loaded pod attached. A pull-up maneuver is then performed by the aircraft until the desired launch conditions (vertical flight path angle, velocity and altitude) are reached for pod release. For a 2 g pull-up maneuver these quantities are as shown by the curves in Figure G-5.

It is obviously desirable to launch the loaded pod from the airplane at high altitude and high velocity. However, as the airplane pulls up, all three quantities (velocity, altitude and flight path angle) are varying. For a typical trajectory* the exchange ratios are

$$\frac{\Delta(\text{payload weight})}{\Delta(\text{launch velocity})} = 0.18 \text{ lb/(ft/sec)}$$

$$\frac{\Delta(\text{payload weight})}{\Delta(\text{launch altitude})} = 1.77 \text{ lb/1000 ft} \qquad \text{(see Figures G-5 and G-6)}$$

No exchange ratio is given for the change of payload weight with launch flight path angle (β_{launch}). This exchange ratio would be meaningful if the angle of attack of the missile with respect to the flight path is the same at the start of powered flight, regardless of the value of β_{launch} (flight path angle). For instance it would be meaningful if the Minuteman angle of attack at launch** (α_{launch}) were always 0 degrees, as in a gravity turn trajectory. For trajectory design 3 the effects of small changes in β_{launch} as the airplane moves along the pull-up curve (Figure G-5) are compensated for by changing the Minuteman α_{launch}.** In effect the Minuteman attitude with respect to the local vertical at launch (ϵ_{launch}) is held constant. The payload weight is very insensitive to changes

*Nominal launch condition: altitude 56,000 feet, velocity 1888 ft/sec, M = 1.95, β = 60 degrees using trajectory design number 3 to inject into a 0.5 percent elliptical orbit at 95 degrees from a 65 nautical mile perigee.

**As soon as the Minuteman control system becomes effective and orients the missile.

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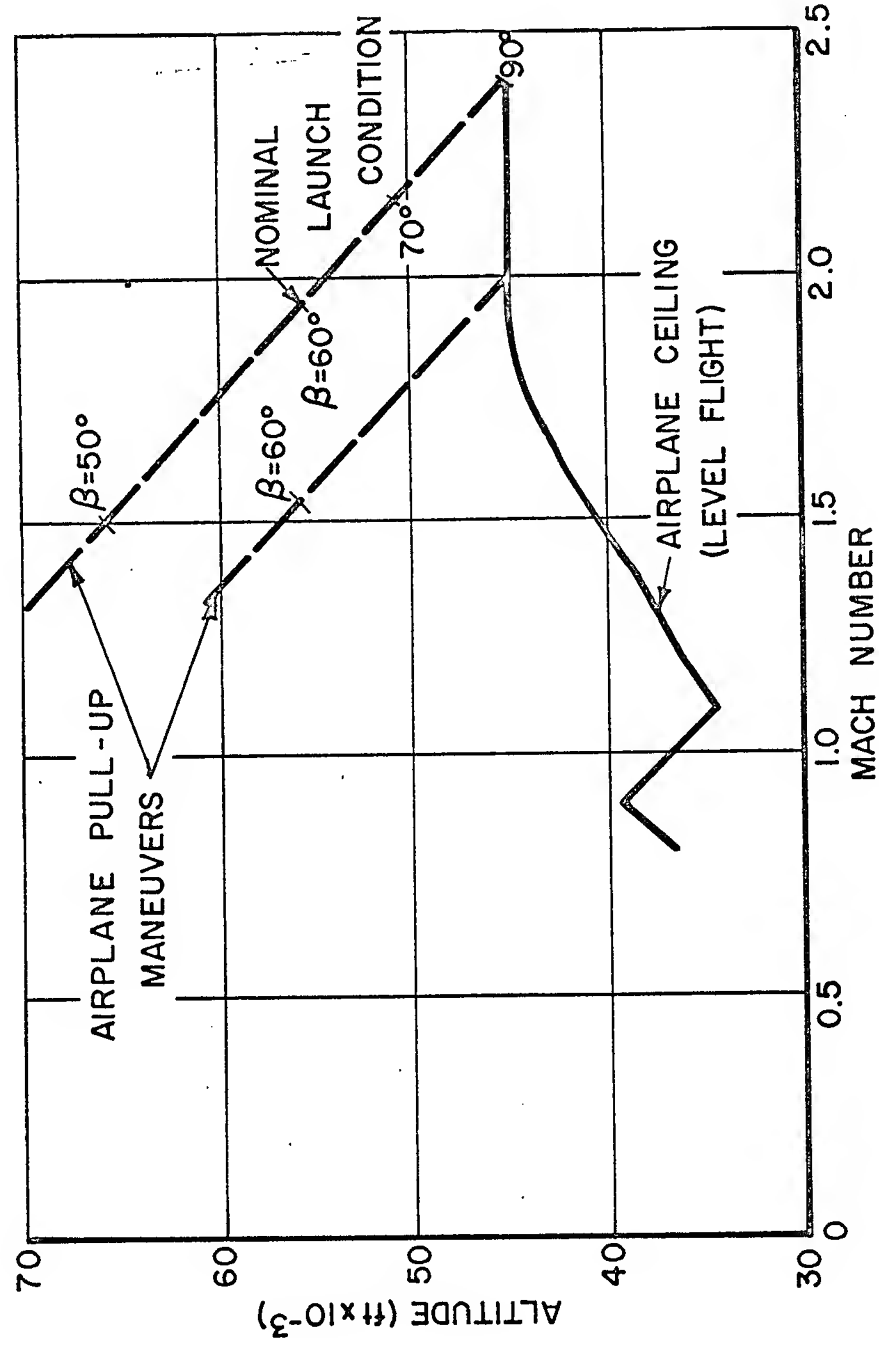

Figure G-5. B-58A Performance for Air Launch Maneuver (70,000 pound pod)

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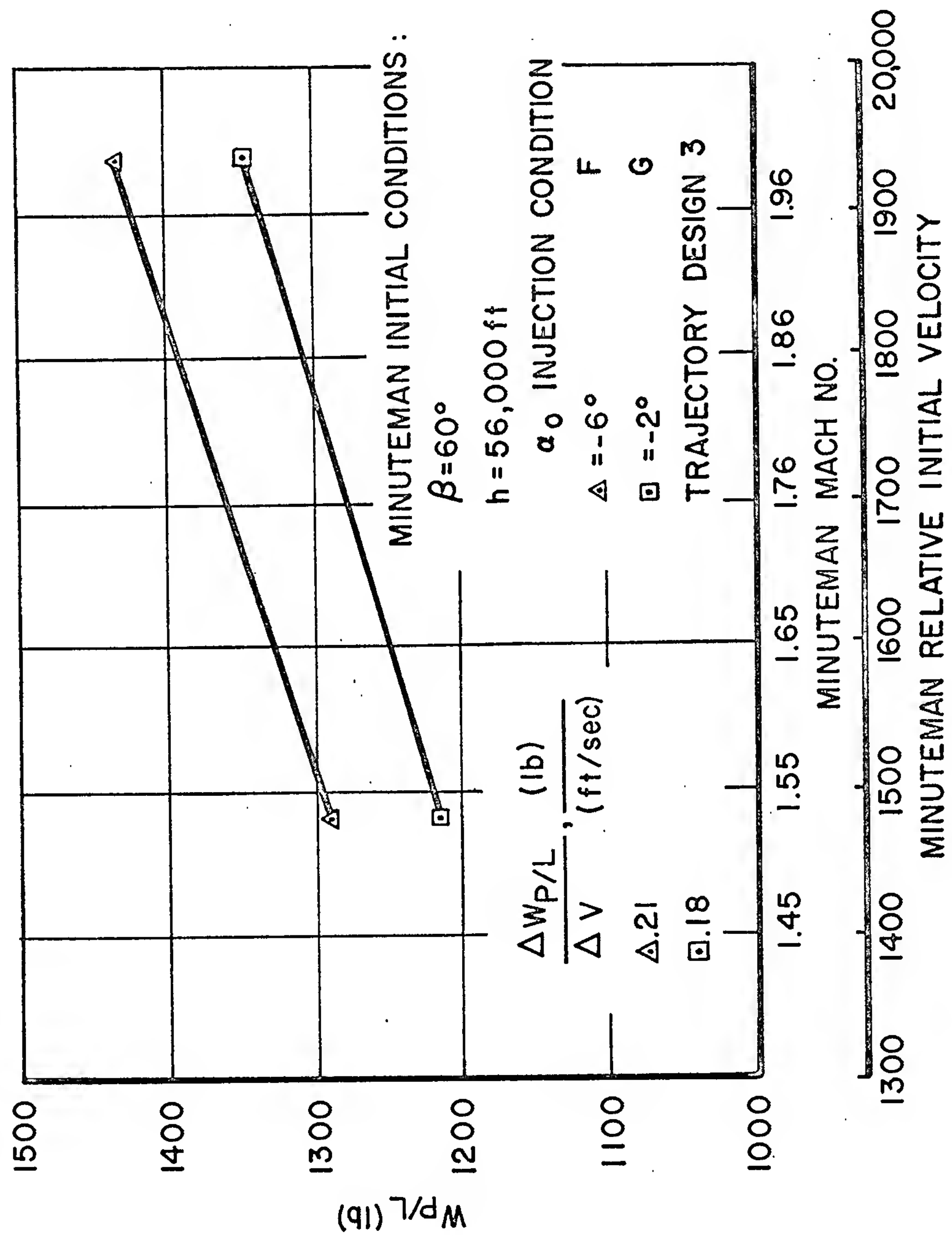

Figure G-6. Minuteman Initial Velocity on Air Launched Minuteman Performance

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in β_{launch} under these conditions. The comparison shown below illustrates the effect:

Minuteman Initial Conditions	Orbit Injection Conditions	Trajectory Design	β_{launch} (deg)	α_{launch} (deg)	ϵ_{launch} (deg)	Payload Weight (lb)
1	G	3	60	-2	62	1345.5
3	G	3	70	+8	62	1339.4*

The payload weight varies only 6.1 pounds for a β_{launch} variation of 10 degrees. In practice this large a change in α_{launch} will not be used. It exceeds the angle of attack limits due to Minuteman control authority limitations but it illustrates the insensitivity of payload weight to changes in β_{launch} when ϵ_{launch} is held constant.

Using these exchange ratios as an indication of the effects of launch velocity and launch altitude at the nominal launch condition, one can get a feel for the effect of small changes in launch conditions on payload weight. For launch conditions shown in Figure G-5, the airplane performance in the 2 g pull-up maneuver shows that as launch altitude increases, launch velocity decreases in the ratio at the rate of 22.6 ft/sec for each 1000 feet altitude increase. The net effect on payload weight for a change in launch altitude of 1000 feet along the 2 g pull-up curve is:

*Corrected for velocity and altitude difference between Minuteman initial conditions 1 and 3.

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	Payload Weight (lb)
Effect of launch velocity change (-22.6 ft/sec)	-4.06
Effect of launch altitude change (+1000 ft)	+1.77
Effect of launch flight path angle change (holding ϵ_{launch} constant)	0
Net effect	-2.29

This indicates the desirability (from a performance standpoint) of launching at lower altitudes and higher velocities in the region of the nominal launch condition studied.

d. Miscellaneous

The basic initial conditions at Minuteman ignition are listed with typical values in Table G-2. In addition to the initial conditions listed, the angle of attack, α_o, which the Minuteman control system would seek at first stage ignition is specified separately for each run.

The Minuteman initial conditions corresponding to the 2 g pull-up from level flight at Mach 2.40 and 45,000 feet altitude were thought to correspond to B-58A maximum performance. It is possible that the B-58A airplane will not exceed Mach 2.0 with its present J-79-5 engines and the loaded pod aboard. Therefore the Minuteman initial conditions may not be typical of the performance of a given version of the B-58A airplane but by using the exchange ratios just discussed, the payload weight in orbit can be very closely approximated for any high altitude supersonic B-58A/MM launch.

In summary, the Minuteman initial conditions are primarily a function of the B-58A performance capability as defined by velocity, altitude and flight

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Table G-2. Minuteman Initial Conditions

Initial Condition	Flight Path Angle,* β (deg)	Altitude, h (ft)	Relative Velocity, v_a (ft/sec)	Mach No.	Velocity Increment due to Earth Rotation (ft/sec)	Description
1	60	56,000	1888	1.95	0 (polar)	Nominal launch condition. Pull-up from level flight at Mach 2.40 and 45,000 ft altitude
2	50	66,000	1500	1.55	0 (polar)	Pull-up from level flight at Mach 2.40 and 40,000 ft altitude with 50 ft/sec velocity increment added
3	70	51,000	2091	2.16	0 (polar)	Pull-up from level flight at Mach 2.40 and 45,000 altitude
4	90	45,000	2323	2.40	0 (polar)	Launch from level flight at Mach 2.40 and 45,000 altitude
5	60	56,000	2638	2.73	0 (polar)	Boost Minuteman 750 ft/sec before first stage ignition, otherwise like initial condition 1
6	50	66,000	2250	2.33	0 (polar)	Boost Minuteman 800 ft/sec before first stage ignition, otherwise like initial condition 27
21	60	56,000	1888	1.95	-700	Like initial condition 1 except launch azimuth is 325° at about 35° N latitude
22	60	56,000	1888	1.95	+700	Like initial condition 1 except launch azimuth is 145° at about 35° N latitude
25 or 26	60	56,000	1481	1.53	0 (polar	Pull-up from level flight at Mach 2.0 and 45,000 ft altitude
27	50	66,000	1450	1.50	0 (polar)	Pull-up from level flight at Mach 2.40 and 45,000 ft altitude

*β Angle between the local vertical and the velocity vector

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path angle at pod release. For launches near the nominal conditions described, and using the B-58A airplane performance curve, Minuteman performance is somewhat improved by launching at lower altitudes because of the increased Minuteman launch velocity. Flight path angle effects (velocity losses due to thrust loss at angle of attack), as discussed in the section on trajectory design, soon override this consideration and limit these payload weight gains.

The method used herein for the calculation of trajectories leads to almost optimum trajectories and is considered suitable for this phase of preliminary information. The Minuteman booster data used in these analyses are shown in Table G-3.

Table G-3. Summary--Minuteman Propulsion System

	Stage I	Stage II	Stage III
OA length inches, maximum	282.65	151.15	82.63
Case OD, inches	65.5	44.3	37.5
Average thrust, pounds, vacuum (nominal)	220,000	51,000	15,000
Action time, seconds, 80°F	58.1	58.0	72.0
Delivered specific impulse, lb-sec, vacuum	268.4	273.5	281.5
Number of nozzles	4	4	4
Nozzle expansion ratio (contour)	10:1	18.4:1	20.4:1
Nozzle exit plane area, square inches	1642.2	1062.0	717.0

3. Orbit Injection Conditions

The Sino-Soviet territory of interest from a reconnaissance standpoint lies between 35° N and 75° N latitudes and spreads out over 180° longitude. To obtain complete photographic coverage between these latitudes it is obvious that orbital inclinations up to 75° must be employed. This requirement affects launch azimuth choices.

The most desirable orbit and therefore, the allowable payload weight is determined by consideration of a number of somewhat conflicting constraints:

a) Near circularity of orbit for uniform altitude throughout the picture-taking region

b) Highly elliptical orbit for longer lifetime

c) Low altitude orbit for maximum ground resolution and maximum payload

d) High altitude orbit to reduce heating and disturbing torques.

The choice of orbit is made by considering the following factors.

For two-dimensional trajectory studies the three orbital injection conditions which must be satisfied at third stage shutdown are altitude, vertical flight path angle, and velocity. The orbit injection conditions prescribed herein are for a spherical nonrotating earth of 3442 nautical mile radius. For the low altitude orbits of small eccentricity considered in this study, the flight path angle is nearly horizontal (β approximately equal to 90°), velocities are high (in the neighborhood of 25,600 ft/sec), and altitude is approximately 105 nautical miles.

Considering these as typical orbital injection conditions, the payload weight in orbit is most strongly affected by injection altitude variations. Figure G-2 shows the effect on payload weight in orbit (using the nominal launch condition and trajectory design 3) of injecting 95° downrange from perigee into circular to one percent elliptical orbits with 65 nautical mile perigee. For these orbits Table G-4 lists the actual values for orbital injection altitude, velocity and flight path angle.

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Table G-4. Range of Injection Parameters for a 65 Nautical Mile Perigee Orbit on a Spherical Earth, Injection 95° Downrange from Perigee

	Circular	One Percent Ellipse
Injection Altitude (n mi)	65	105
Injection Velocity (ft/sec)	25,713	25,563
Injection Angle (β)	90°	89.42°

The orbital injection velocity tradeoff factor is:

$$\frac{\Delta(\text{Payload Weight})}{\Delta(\text{Injection Velocity})} = -0.254 \frac{\text{lb}}{(\text{ft/sec})}$$

This trade-off factor yields a net increase in payload weight of 38 pounds due to a decrease in required injection velocity from a circular orbit to a one percent ellipse.

No reliable trajectory data is yet available to check the effect on payload weight of varying only the vertical flight path angle holding injection velocity and altitude constant. The gain in payload weight due to flight path angle decrease between injecting into a circular orbit and a one percent elliptical orbit is estimated to be less than 25 pounds. The accumulative effect going from circular to one percent elliptical orbits due to changes in the injection velocity and vertical flight path angle is less than 65 pounds gain in payload weight. Figure G-2 shows a 370-pound decrease in payload weight between injecting into the circular and one percent elliptical orbits, resulting in the conclusion that the net effect of changing injection altitude from 65 nautical miles to 105 nautical miles is approximately 433 pounds decrease in payload weight.

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Table G-5 presents the data on orbital injection conditions which were used for this study. These injection conditions correspond to idealized orbits over a spherical earth. More realistic injection conditions are represented by the orbital parameters for the osculating ellipse traced out by the low altitude satellite over an ellipsoidal earth. The primary effect on the payload weight made by using the osculating elliptical path to calculate orbital injection conditions is the change in injection altitude which results (see Figure H-1 to H-7) in Appendix H). In general for these highly inclined orbits, the injection altitude will be lower by up to 10 nautical miles than would be computed for the idealized case. This effect can increase payload weights by as much as 100 pounds.

The 65 nautical mile altitude perigee was picked in order to get the satellite as low over the target as possible in order to keep photographic resolution high. It now appears that a 65 nautical mile perigee is slightly too low an altitude for picture taking because of aerodynamic heating of the outside camera element at that altitude. The perigee altitude is now more likely to be at 70 nautical miles for the nominal trajectory. However, this change will not seriously affect the payload weight orbited since injection occurs at 95° downrange from perigee and only small changes in injection conditions would result from raising the perigee, assuming that the apogee were lowered a corresponding amount. For a specific orbital injection condition, the orbited payload weight may be approximated using the trade-off factors previously stipulated.

4. <u>Trajectory Design</u>

The objective in trajectory design is to maximize the payload weight by optimizing the Minuteman flight program, i.e., the flight path starting with the Minuteman initial conditions and ending with orbital injection conditions.

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Table G-5. Orbital Injection Conditions

Injection Condition	Injection Velocity (ft/sec)	Injection Altitude (n mi)	β (deg)	Orbital Ellipticity	Perigee Altitude (n mi)	Injection-Perigee Separation (deg)	Notes
A	25,355	128	89.70	0.01	65	150	Never used
B	25,563	105	89.42	0.01	65	95	
C	25,713	65	90	0	--	--	Circular 65 n mi orbit
D	25,640	85	90	0	--	--	Circular 85 n mi orbit
E	25,568	105	90	0	--	--	Circular 105 n mi
F	25,671	75	89.86	0.0025	65	95	
G	25,635	85	89.72	0.005	65	95	
H	25,599	95	89.57	0.0075	65	95	
I	25,618	82.3	90.00	0.0025	65	180	Injection at apogee
J	25,519	100.1	90.00	0.005	65	180	Injection at apogee

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Along the flight path certain constraints must be satisfied. For example, the angle of attack of the Minuteman must be maintained within control limits during flight in the atmosphere. Such constraints as inertial platform maximum turning rates and time required for data smoothing before third stage engine cutoff may impose further constraints on the trajectory design, if inertial guidance is used. For radio guidance, look angles also impose limits.

Figure G-7 shows typical velocity losses for Minuteman trajectory design 3 injecting the payload into orbit at 75 nautical miles altitude. The total velocity loss is equal to the relative velocity subtracted from the ideal velocity, where ideal velocity is that attainable from the total impulse applied by the Minuteman engines if no losses occurred. Total velocity loss is also equal to the sum of the velocity losses due to gravity, thrust losses due to angle of attack, drag losses, and atmospheric thrust losses. The payload weights corresponding to these trajectories are plotted in Figure G-1. Of course the maximum payload will correspond to minimum velocity losses for a given trajectory.

The velocity losses due to aerodynamic drag and to atmospheric thrust loss are both small for these trajectory designs and increase only slightly as the Minuteman initial angle of attack is lowered. Figure G-8 shows the flight paths for these trajectories and helps in visualizing the trajectories when considering losses. The high altitude at Minuteman first stage ignition explains the low value of these velocity losses (compared to ground launched losses) encountered due to the presence of the atmosphere.

The loss due to gravity is the largest of the four velocity losses. It decreases slowly as the trajectory flattens out. The gravity losses are considerably lower for these trajectories than would be encountered in a vertical

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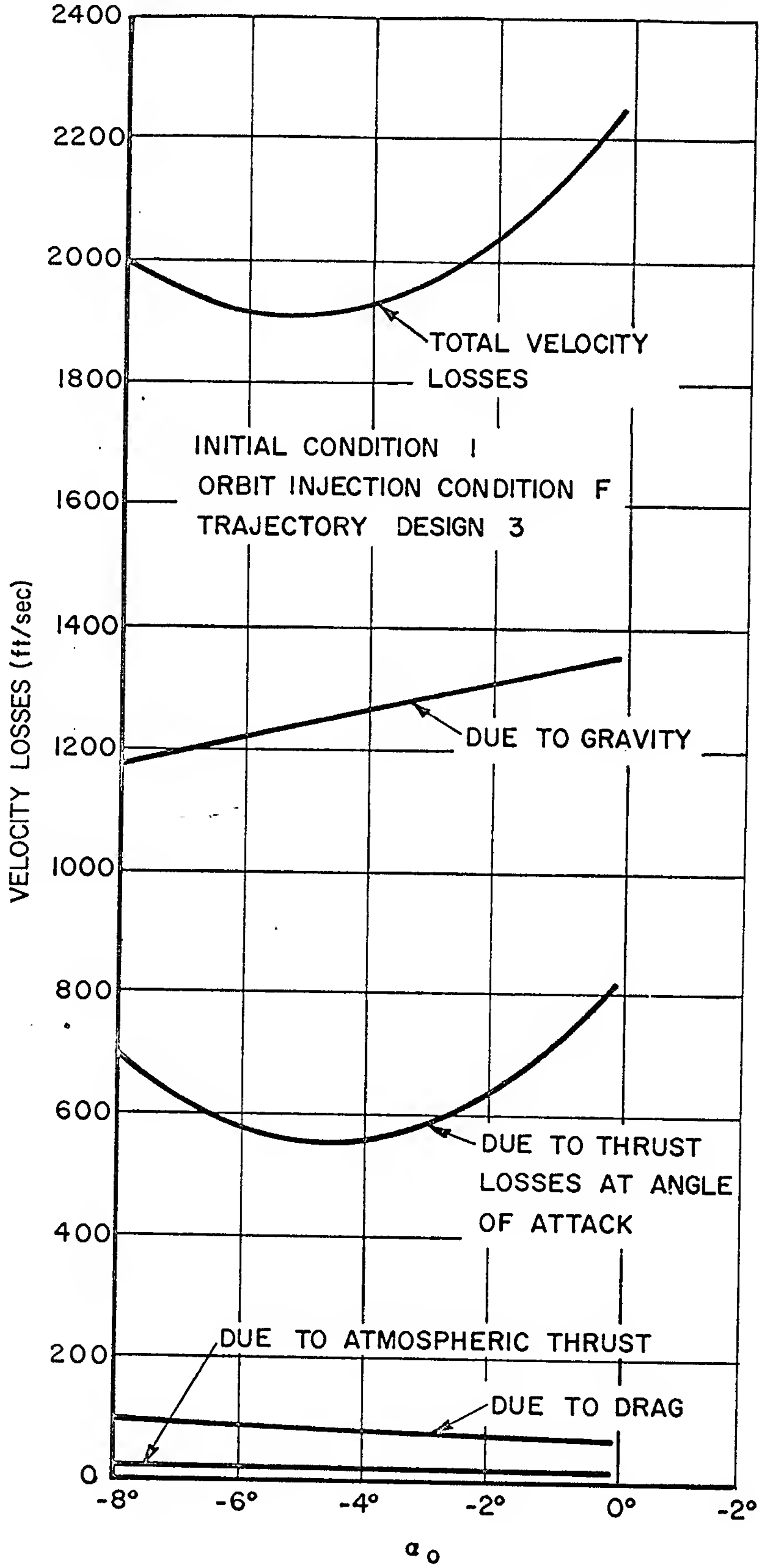

Figure G-7. Typical Air Launch Minuteman Velocity Losses

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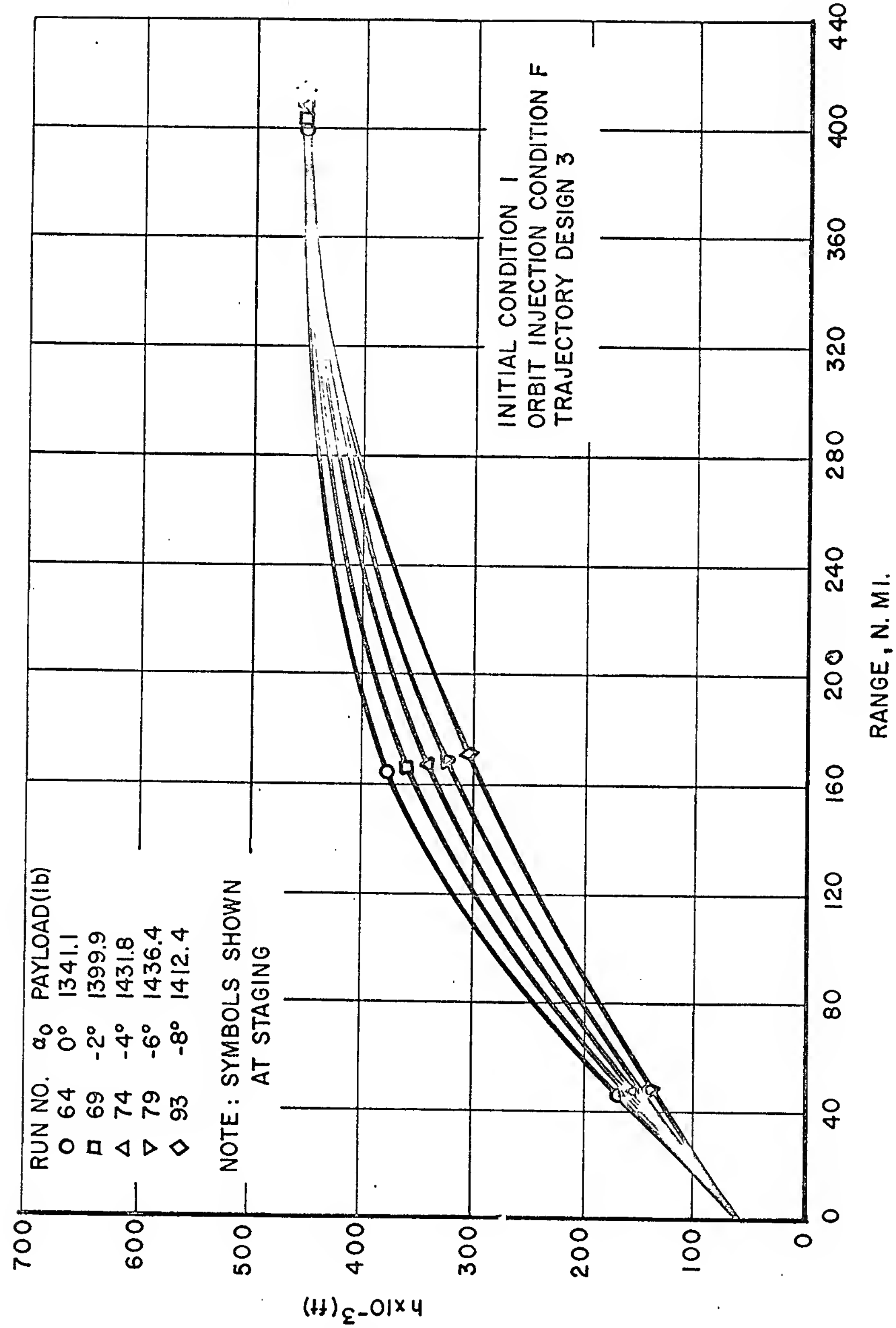

Figure G-8. Air Launch Minuteman Powered Flight Paths

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launch. The velocity losses due to gravity are proportional to cos β. For a ground launch, β is normally 0° at first stage ignition and remains less than 60° for a majority of the first stage burning period. For the air launched trajectories β at first stage ignition is approximately 60°.

The velocity loss due to thrust loss at angle of attack can also be described as the loss due to the thrust vector not being aligned to the velocity vector. This loss becomes really significant whenever the angle of attack (α) increases much beyond 10 degrees, since the loss is approximately proportional to sin α. For this trajectory the increase in velocity loss due to thrust loss at the $\alpha_o = -8^{\circ}$ end of the curve is due to the larger negative angle of attack required in the third stage burn period in order to force the velocity vector down to a near horizontal attitude for orbit injection. The increase in the velocity losses due to thrust at the $\alpha_o = 0^{\circ}$ end of the curve is due to the larger angles of attack encountered in the early part of the trajectory in lofting the flight path.

The mathematical statement for the calculation of the velocity losses on the electronic computer is:

$$V_d = \int \frac{g_1 D}{W} dt$$

$$V_g = \int \frac{g_o \cos \beta}{\sigma^2} dt$$

$$V_\alpha = \frac{g_1 F}{W} dt - V_r - V_g - V_d$$

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$$V_a = V_{ideal} - \int \frac{g_1 F}{W} dt$$

$$V_T = V_{ideal} - V_r$$

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V_d = Velocity loss due to aerodynamic drag

V_g = Velocity loss due to gravity

V_α = Velocity loss due to thrust loss at angle of attack

V_a = Velocity loss due to atmospheric thrust loss

V_T = Total velocity loss (ft/sec)

V_r = Relative velocity

V_{ideal} = Ideal velocity (total impulse velocity)

g_1 = Sea level acceleration due to gravity (32.174 ft/sec^2)

g_o = Gravitational constant (1.4071613×10^{16} ft^3/sec^2)

σ = Radius vector from the center of the earth

F = Thrust

W = Weight

t = Time

β = Angle between local vertical and velocity vector

For this Minuteman performance feasibility study several trajectory designs were investigated for the purpose of selecting one which is reasonably close to optimum and suitable for use in this study. A few remarks about the trajectories tried (see Table G-6) gives some insight into the trajectory design problem.

For a given set of initial conditions the STL two-dimensional trajectory simulation program used in this performance study computes the correct values for two trajectory parameters (e.g., pitch rate or kick angle) and the payload weight to satisfy the three orbit injection conditions (attitude, flight path angle and velocity). These features make it a powerful tool for rapidly trying and selecting trajectory designs for feasibility study purposes. The results of simulating the various trajectory designs listed in Table G-2 will be discussed only briefly.

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Table G-6. Trajectory Designs

Trajectory Design Number	Initial Kick	First Stage Operation	Second Stage Operation	Third Stage Operation	Notes	Trajectory Description
1	0	Gravity Turn	$\dot{\epsilon}$ = constant*	$\dot{\epsilon}$ = constant*	$\dot{\epsilon} \sim$ pitch rate	First stage gravity turn
2	Computed*	$\dot{\epsilon}$ = constant*	$\dot{\epsilon} = \dot{\epsilon}$ for stage one	$\dot{\epsilon} = 90^{\circ}$	Trajectory did not run $\epsilon \sim$ angle with respect to the launch vertical	Third stage attitude near horizontal, pitch rates equal first and second stage
3	Prescribe	$\dot{\epsilon}$ = 0	$\dot{\epsilon}$ = constant*	$\dot{\epsilon}$ = constant*	Prescribe $\sim$ prescribe as an input to the simulation	Minuteman aimed at first stage ignition
5	0	$\dot{\epsilon}$ = constant*	$\dot{\epsilon}$ = constant*	$\dot{\epsilon} = 90^{\circ}$		Third stage burn attitude
7	0	$\dot{\epsilon}$ = constant*	$\dot{\epsilon}$ = constant*	$\dot{\epsilon}$ = constant		Third stage constant attitude
8	Prescribe	$\dot{\epsilon}$ = constant*	$\dot{\epsilon}$ = constant*	$\dot{\epsilon}$ = constant		Minuteman aimed at first stage ignition, third stage constant attitude
9	Prescribe	$\dot{\epsilon}$ = constant*	$\dot{\epsilon}$ = constant*	$\dot{\epsilon}$ = prescribe		Used to explore performance improvement possibilities by pitching first stage on trajectories similar to trajectory design 3
10	Prescribe	$\dot{\epsilon}$ = 0.01215	$\dot{\epsilon}$ = constant*	$\dot{\epsilon}$ = constant*	$\dot{\epsilon}_1$ = earth curvature rate trajectory specified to match 3D trajectory for checking purposes	
11	Prescribe	$\dot{\epsilon}$ = 0	$\dot{\epsilon}$ = constant*	$\dot{\epsilon}$ = constant*	Prescribe coast $(t, \dot{\epsilon})$ after second stage burnout	Coast trajectory otherwise similar to trajectory design 3
12	Computed*	$\dot{\epsilon}$ = 0	$\dot{\epsilon}$ = 0	$\dot{\epsilon}$ = constant*	Prescribe coast $(t, \dot{\epsilon})$ after second stage burnout	Alternate coast trajectory
13	Prescribe	Gravity Turn	$\dot{\epsilon}$ = constant*	$\dot{\epsilon}$ = constant*		First stage gravity turn at selected initial flight path angle

*These quantities were computed as a part of the trajectory calculation in order to meet the requirement for injection at third stage burnout.

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First stage gravity turn trajectories (see trajectory designs 1 and 13) have the advantage of keeping the Minuteman angle of attack near zero during atmospheric flight, thereby minimizing control system problems due to aerodynamic instabilities. The reason for running these trajectories was to evaluate the associated payload weight penalties. When compared to payloads obtained for trajectory design 3, the payload weight penalty for the gravity turn trajectories was a minimum of 100 pounds.

Trajectory design 2 proved to be impractical. It is a highly lofted trajectory. The two times it was tried, the machine program could not calculate a trajectory because it could not turn the velocity vector far enough to get it near horizontal ($\beta = 90^{\circ}$) at injection.

Trajectory design 3 proved to be very effective for use in the feasibility study. Payload weights are maximum when compared to other trajectory designs simulated as a part of this study. For a given set of orbit injection conditions, the launch (initial) conditions (along the airplane performance curve) which will maximize payload weight are readily found (see Figure G-1). This figure also shows that the Minuteman performance capability is quite sensitive to ϵ (missile angle with respect to launch vertical). This is the reason for labeling this trajectory "aimed." The maximum payload is obtained by aiming the Minuteman in the direction which will enable it to follow a flight path which is as flat as possible, but still keep thrust required for turning the velocity vector to near horizontal to a minimum (see discussion of velocity losses and Figures G-7 and G-8).

Trajectory design 5 was run to evaluate the concept of using the third stage thrust only to gain velocity and thereby force all the flight path direction turning to be done by Stages 1 and 2. This proved to be an impractical trajectory because of the high pitch rate required during second stage burning. Payload weights were very small for these trajectories.

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Trajectory designs 7 and 8 were the first trajectories run in the study and resulted in some of the early payload weight results obtained. The payload weights resulting from trajectory design 3 were higher by 40 to 65 pounds and the latter was used in this performance study.

Trajectory design 9 was designed to investigate the possibility of improving on trajectory design 3 payload weights by incorporating a first stage pitch rate into the trajectory. Four exploratory trajectory simulations were made. The data is not presented in the report since more work is required to obtain conclusive results. However these trajectories do diminish the sensitivity of payload weight to ϵ (the Minuteman angle with respect to the launch vertical) at Minuteman ignition. Work with this trajectory was discontinued because of the time restrictions associated with the difficultites of optimizing the pitch rates on all three stages for each trajectory. There is much work which could be done with this and other trajectory designs.

Trajectory design 10 was used solely to check payload weight results from the two-dimensional simulation being discussed here with the results of the three-dimensional trajectory simulation (see section on three-dimensional simulation program results).

Trajectory designs 11 and 12 were set up to evaluate performance using a trajectory with a coast period between second stage burnout and third stage ignition. First results from these trajectory simulations indicate that substantial payload weight increases are possible using coast trajectories. Using trajectory design 11, with a coast period of 100 seconds, no coast pitch rate, initial condition 1, and orbit injection conditions B, a payload weight in orbit

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of 1390 pounds was obtained. This is an increase of 240 pounds over the best payload weight obtained for the same Minuteman initial conditions and orbit injection conditions using continuous burn trajectories (trajectory design 3). Much work remains to establish the optimum coast trajectory designs and effects on performance for various launch and orbital injection conditions. The effects of the coast period on the Minuteman operation also remain to be evaluated (e.g., attitude control requirements during coast, effects on Minuteman staging, etc.).

5. Three-Dimensional Trajectory Simulation Studies

The IBM computer program for simulating a three-dimensional (3-D) powered flight trajectory was used mainly as a means of corroborating the payload capability results obtained by a two-dimensional (2-D) program. The 3-D program consumes considerably more computer time than the 2-D program, so that for feasibility studies the relatively simple powered flight program of the 2-D program is preferred. The 3-D program is advantageous when one or more of the following conditions is important.

a) The powered flight may be complicated by one or more coast periods, or by yaw maneuvers

b) Precise geographical location of injection or perigee, etc., is required

c) Radar visibility or special guidance constraints may exist

d) A complicated propulsion simulation of one or more stages may be required

e) Exact simulation of the earth's rotation or the earth track on a nonspherical earth is desired.

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For this configuration where the stages burn without interruption and the true anomaly between launch and injection is only 6 degrees, the 2-D program is considerably more efficient for examining the feasibility of various launch and injection conditions.

The following conditions were used for computing 3-D trajectories to compare with 2-D calculations (for polar orbits):

Injection (Condition B)

a) altitude = 105 n mi

b) inertial velocity = 25,563 ft/sec

c) flight path angle = 89.42 down from local vertical

Launch (Initial Condition 6)

a) altitude = 66,000 feet

b) velocity (relative to air) = 2250 ft/sec

c) flight path angle = 50° down from local vertical

(Initial Condition 2)

a) altitude = 66,000 feet

b) velocity (relative to air) = 1500 ft/sec

c) flight path angle = 50° down from local vertical

For Minuteman initial condition 6 the 3-D program computed a payload of 1303 pounds using the following pitch rate program:

a) Launch thrust axis = 50° from vertical

b) Stage one, gravity turn

c) Stage two, 0.29 deg/sec

d) Stage three, 0.65 deg/sec

The comparable payload from 2-D calculations was 1305 pounds.

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For Minuteman initial condition 2 the 3-D program computed a payload of 1062 pounds using the following pitch rate program:

a) Launch thrust axis = 55° from vertical

b) Stage one, thrust axis = 55° from local vertical (constant)

c) Stage two, 0.89 deg/sec

d) Stage three, 0.08 deg/sec

The comparable payload from 2-D calculations was 1058 pounds.

An alternate computation yielded 1096 pounds using the following pitch rate program:

a) Launch thrust axis = 50° from vertical

b) Stage one, 0.30 deg/sec

c) Stage two, 0.35 deg/sec

d) Stage three, 0.60 deg/sec

These computations may require some modification when more data are known concerning the angle of attack limitations. In general, gravity turns during the first stage (for minimizing aerodynamic loads, pitching moments, and bending moments) tend to reduce payload capability for these launch conditions because of insufficient gains in altitude during first stage burning. Hence, the above pitch programs use less nose-down pitch rates than a gravity turn requires, and this in turn produces nonzero angles of attack during the first stage operation occurring in the atmosphere.

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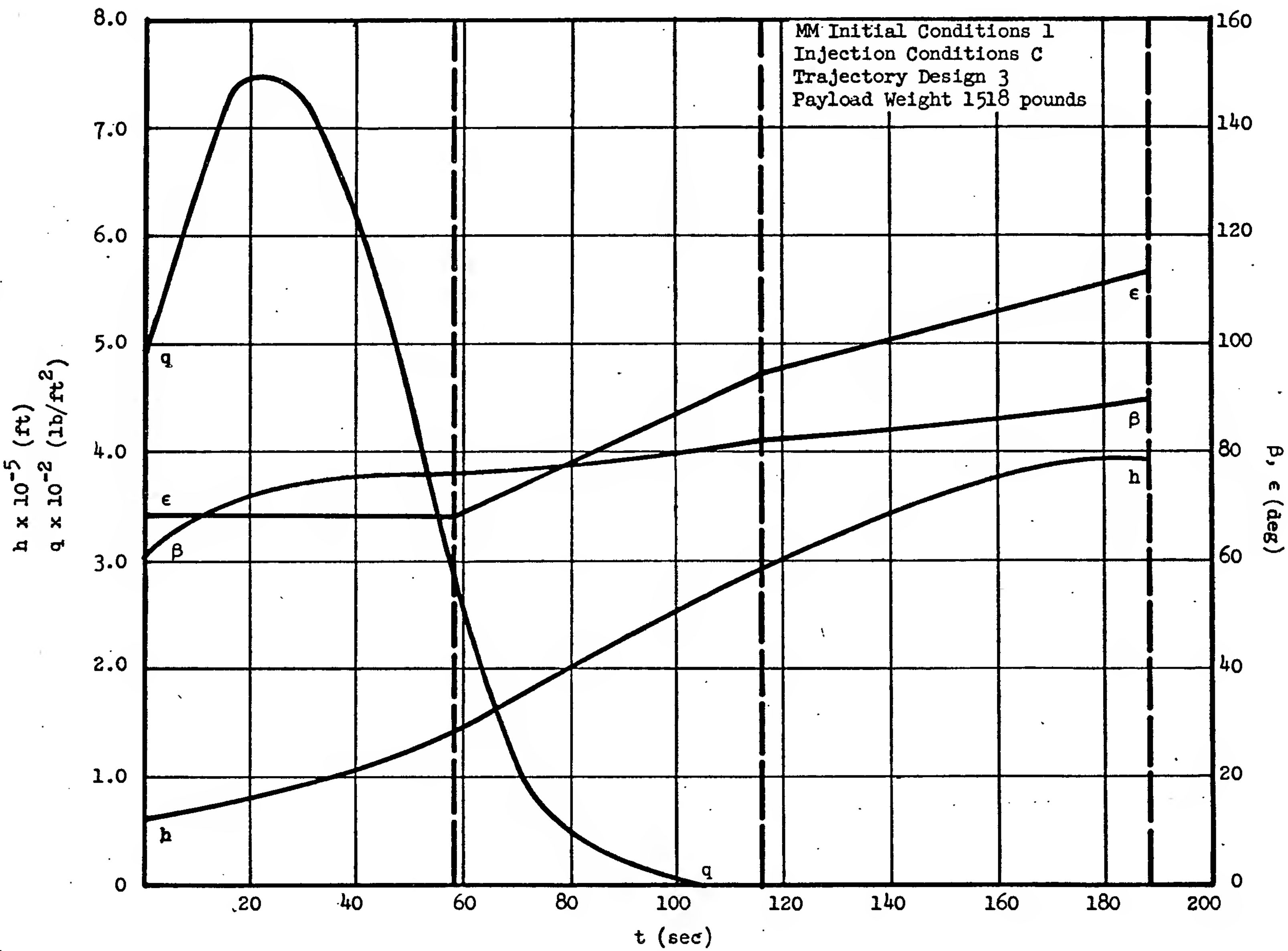

Figure G-9. Reference Trajectory for Orbital Injection Conditions C (circular)

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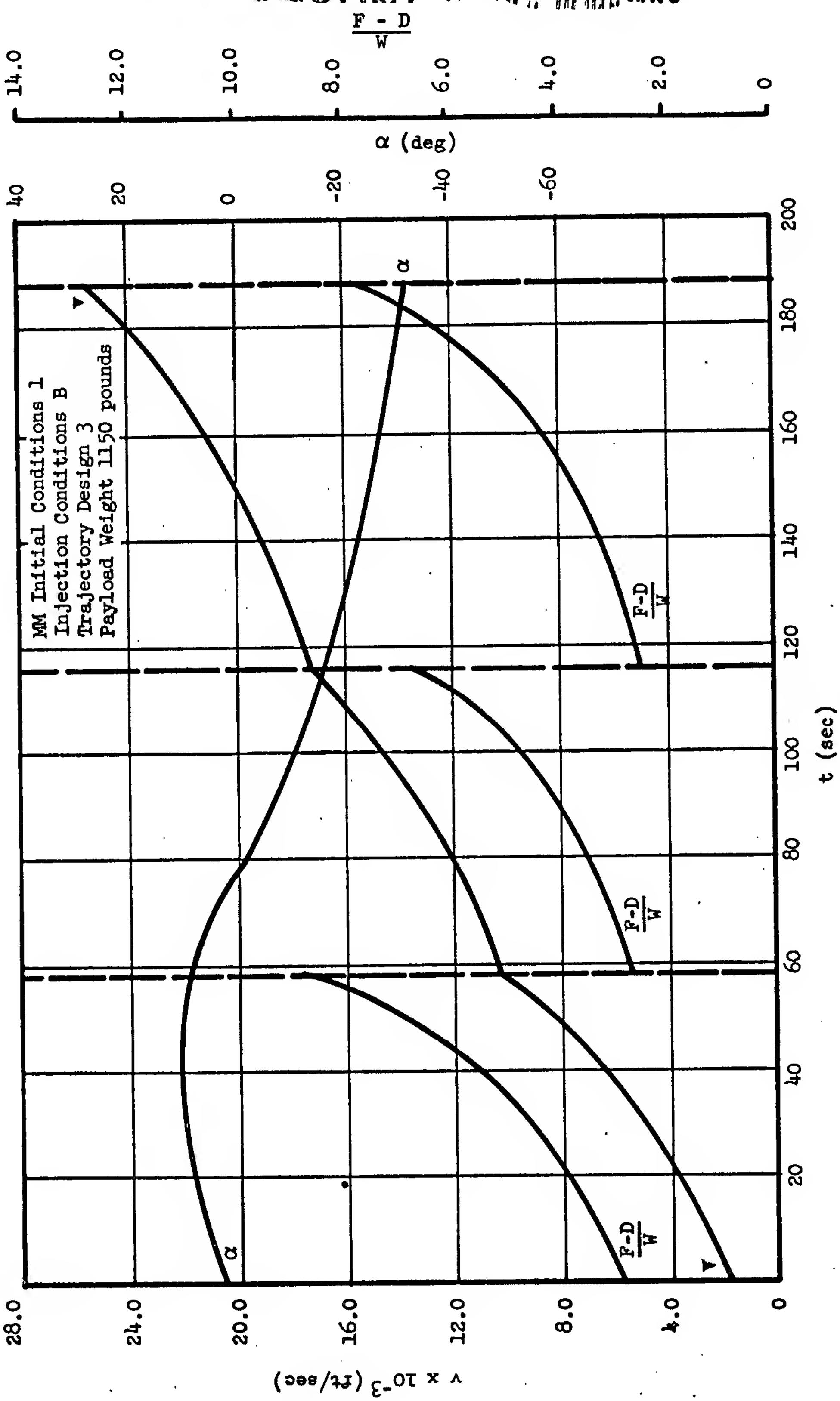

Figure G-10. Reference Trajectory for Orbital Injection Conditions C (circular)

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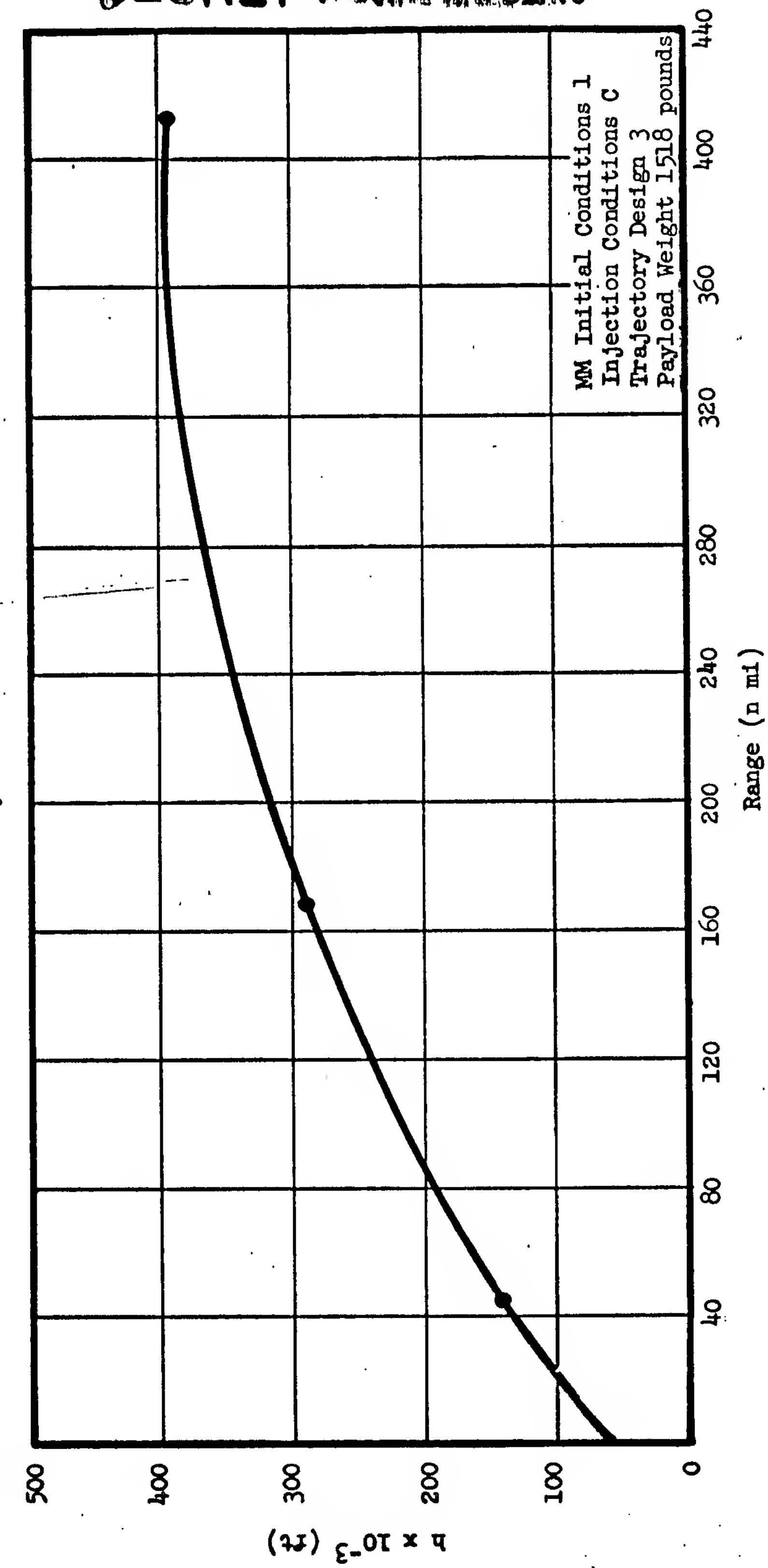

Figure G-11. Reference Trajectory for Orbital Injection Conditions C (circular)

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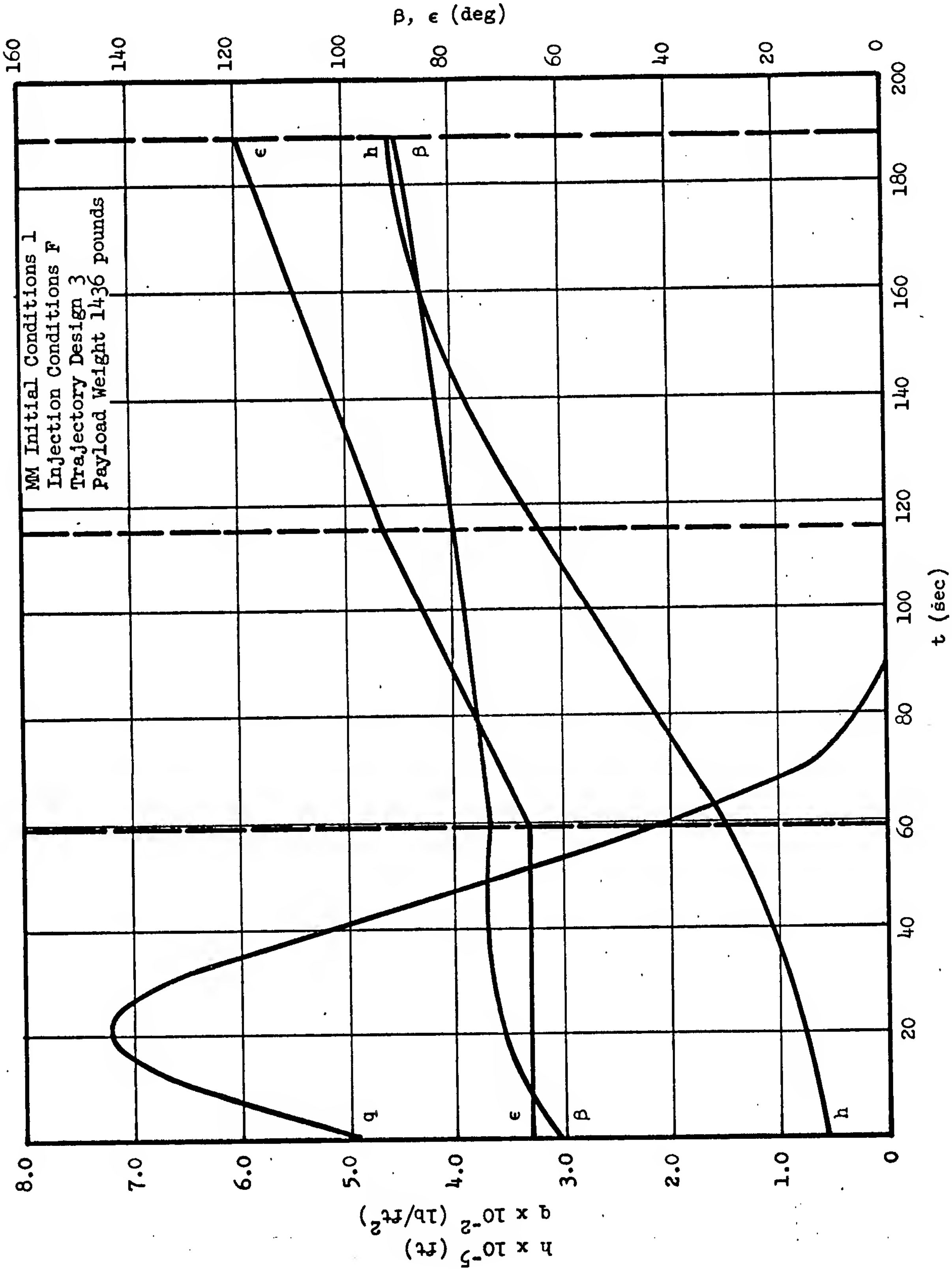

Figure G-12. Reference Trajectory for Orbital Injection Conditions F (1/4 percent ellipse)

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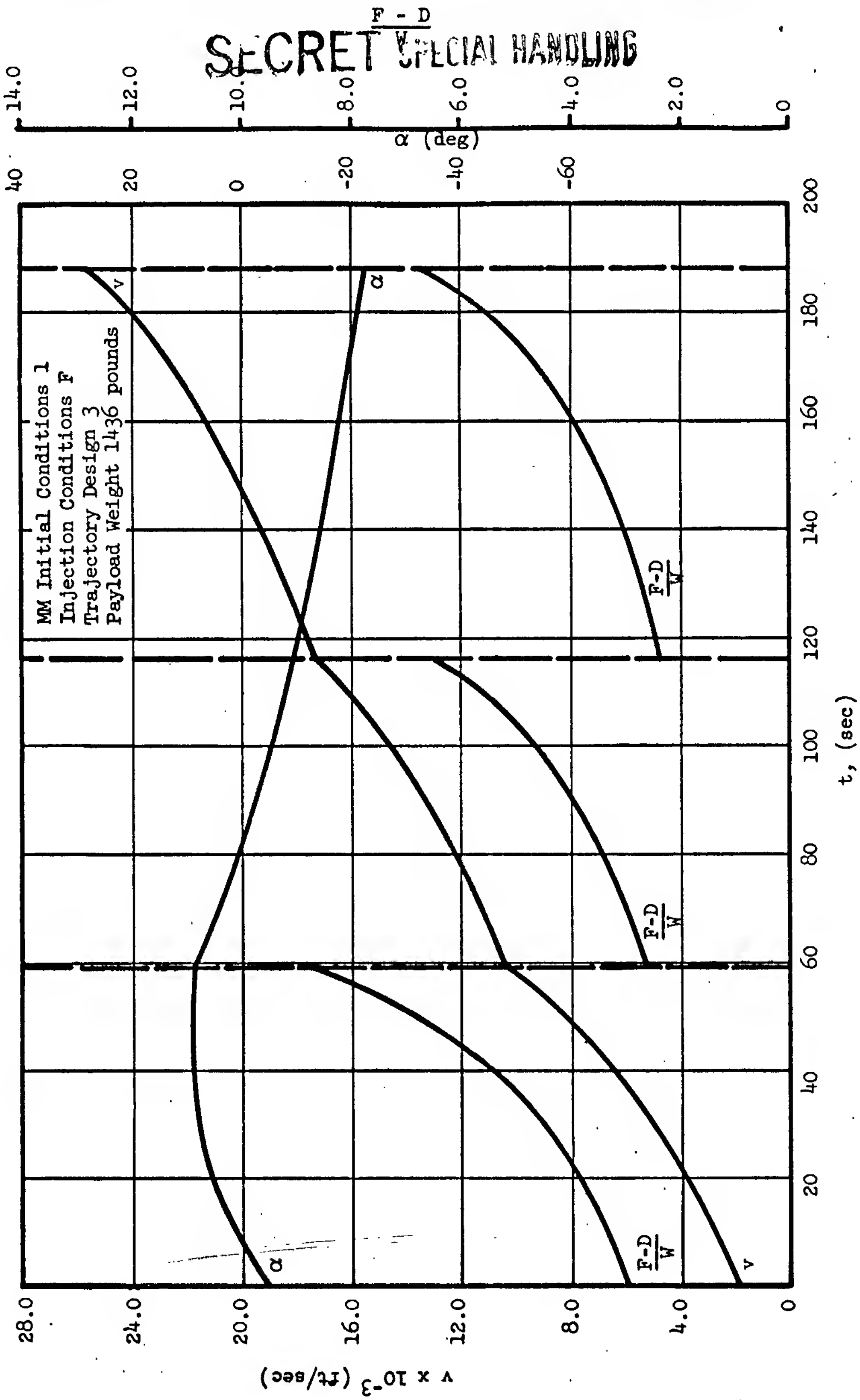

Figure G-13. Reference Trajectory for Orbital Injection Conditions F (1/4 percent ellipse)

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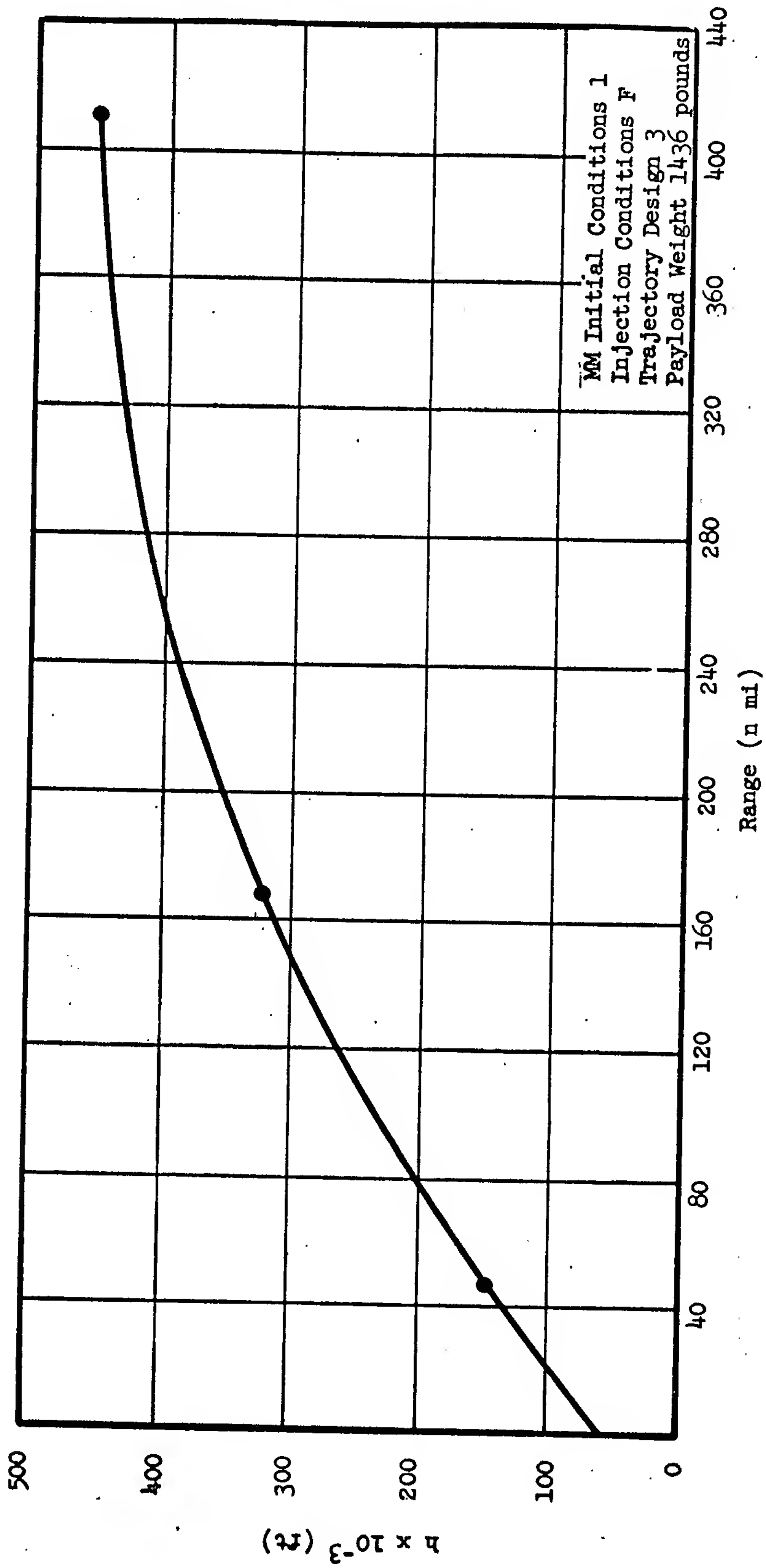

Figure G-14. Reference Trajectory for Orbital Injection Conditions F (1/4 percent ellipse)

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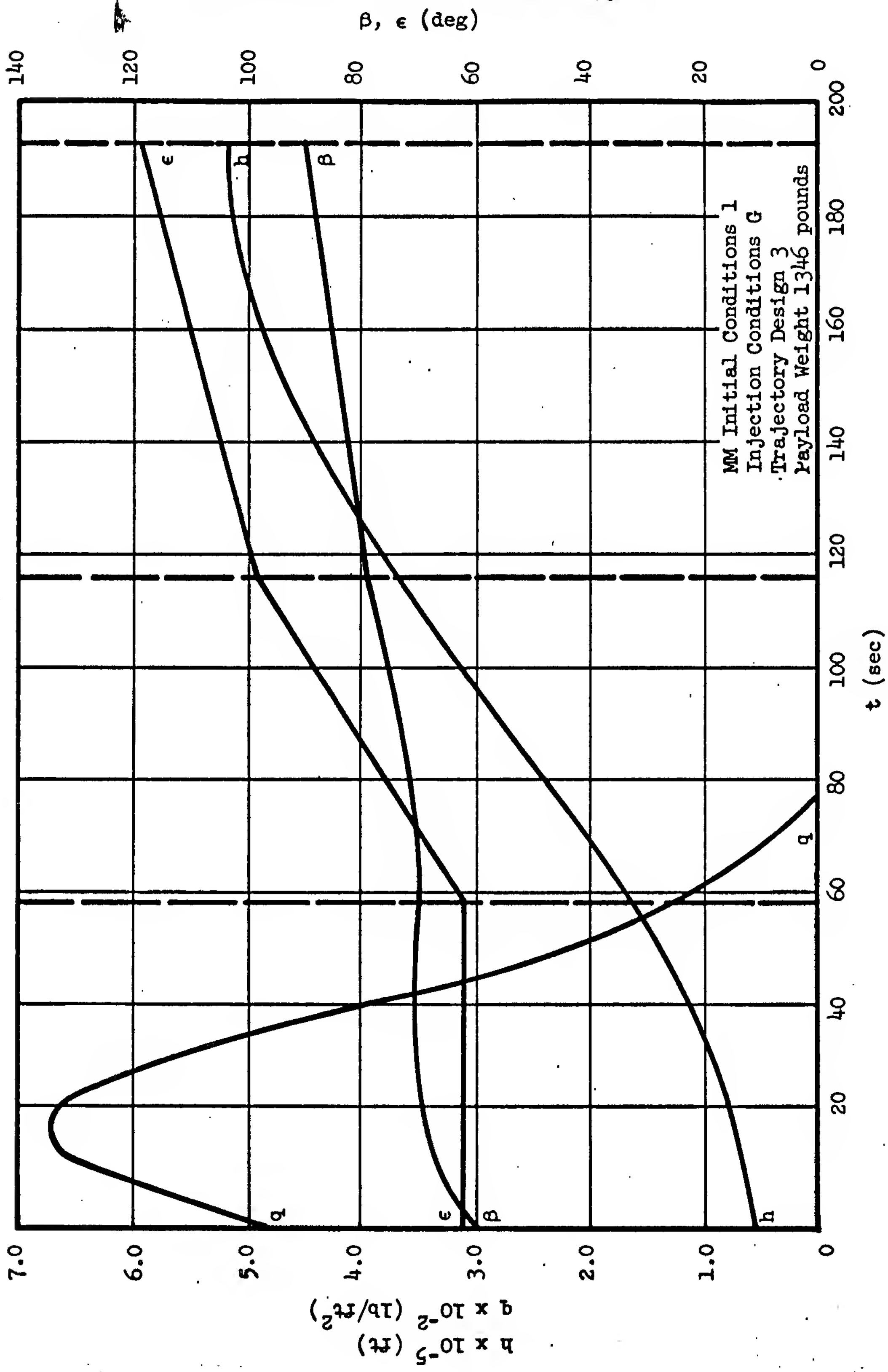

Figure G-15. Reference Trajectory for Orbital Injection Conditions G (1/2 percent ellipse)

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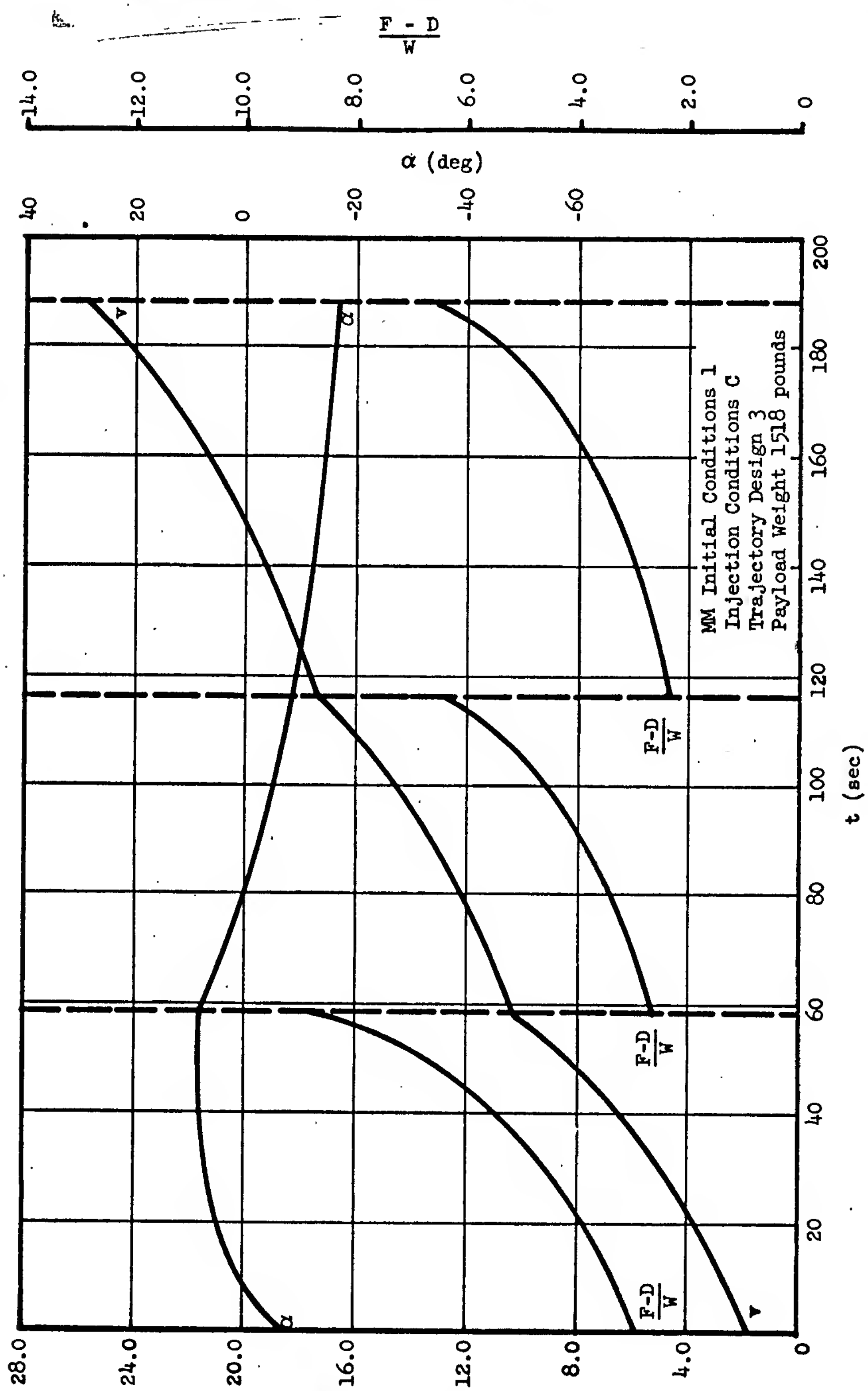

Figure G-16. Reference Trajectory for Orbital Injection Conditions G (1/2 percent ellipse)

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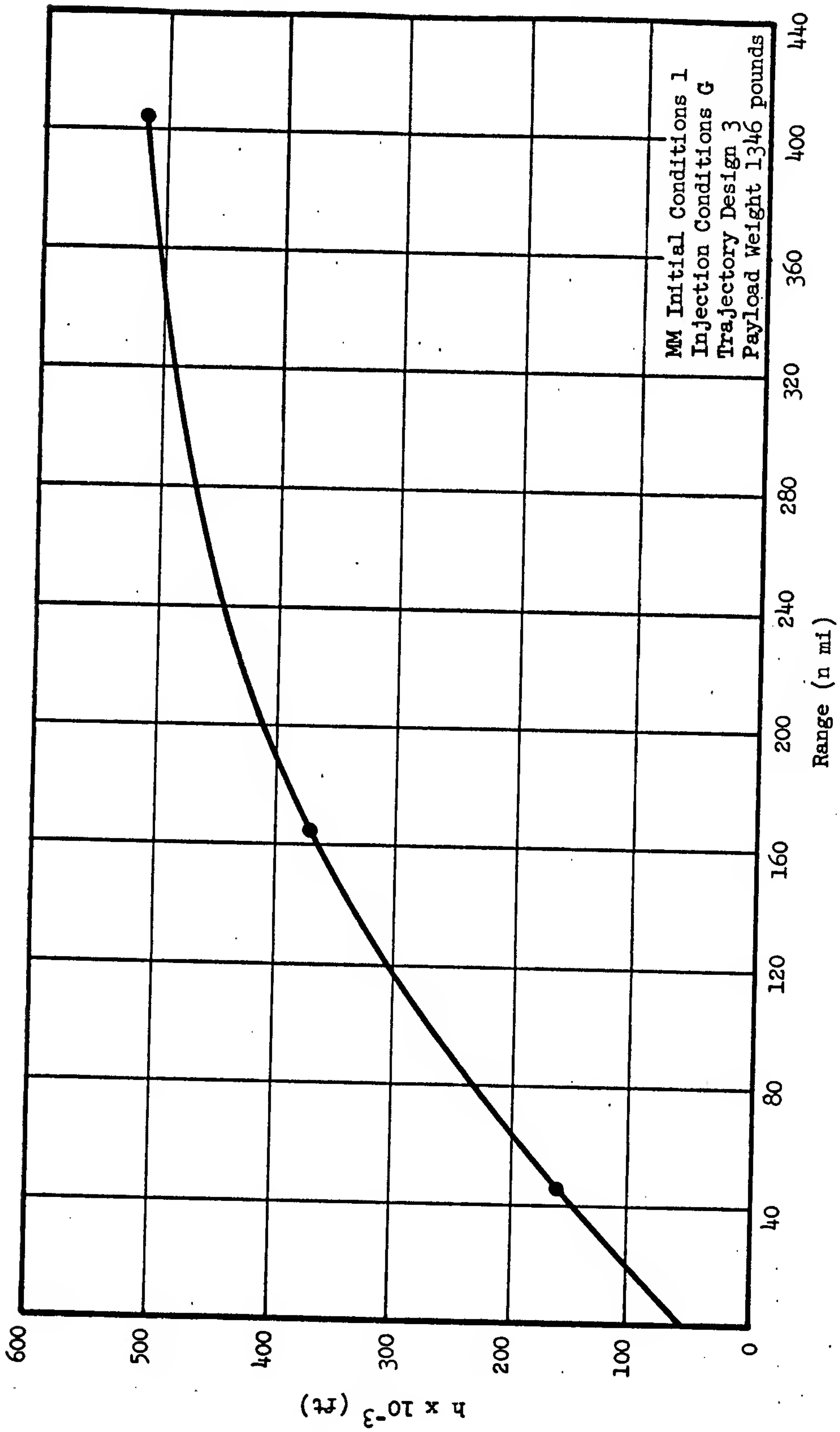

Figure G-17. Reference Trajectory for Orbital Injection Conditions G (1/2 percent ellipse)

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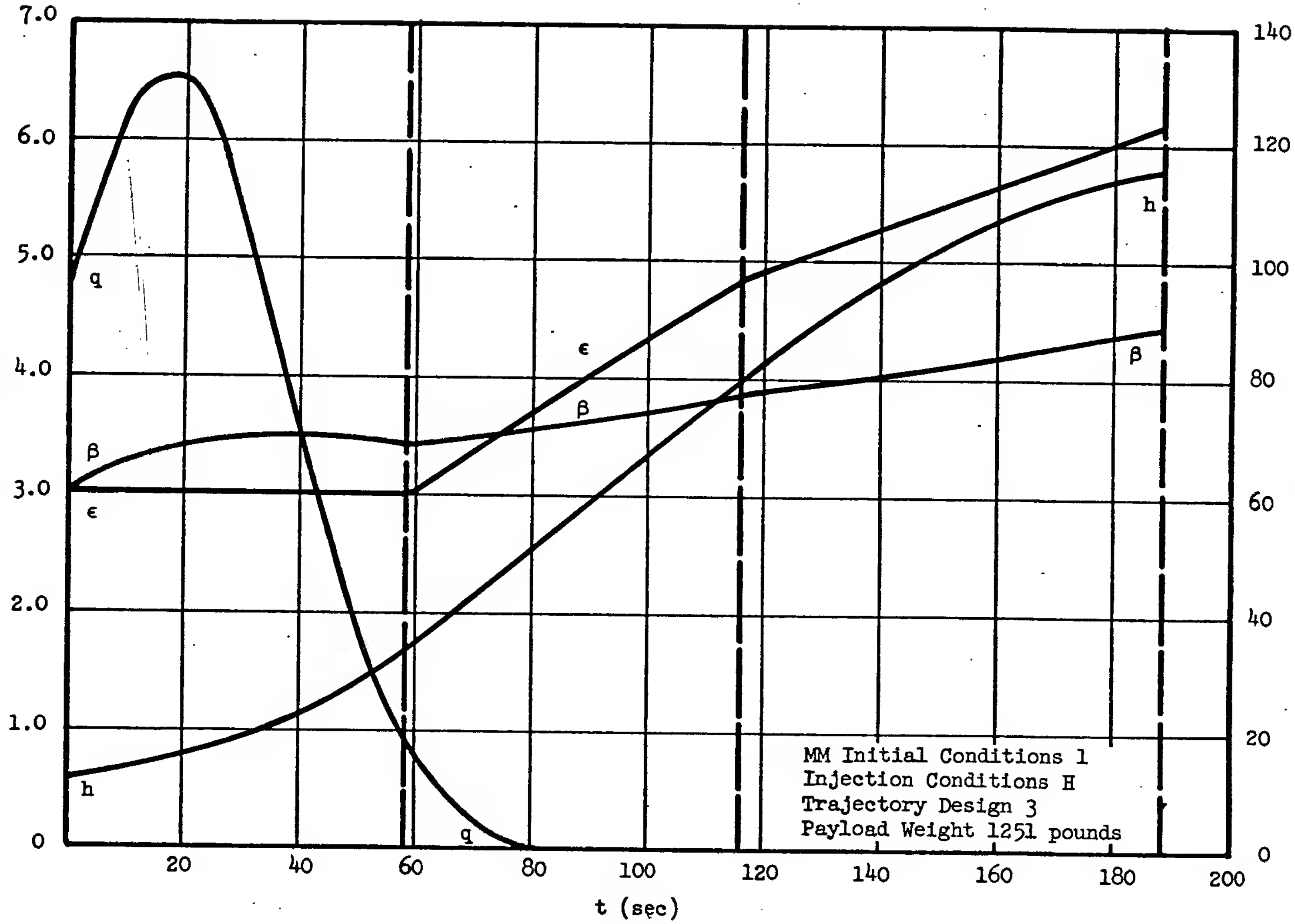

Figure G-18. Reference Trajectory for Orbital Injection Conditions H (3/4 percent ellipse)

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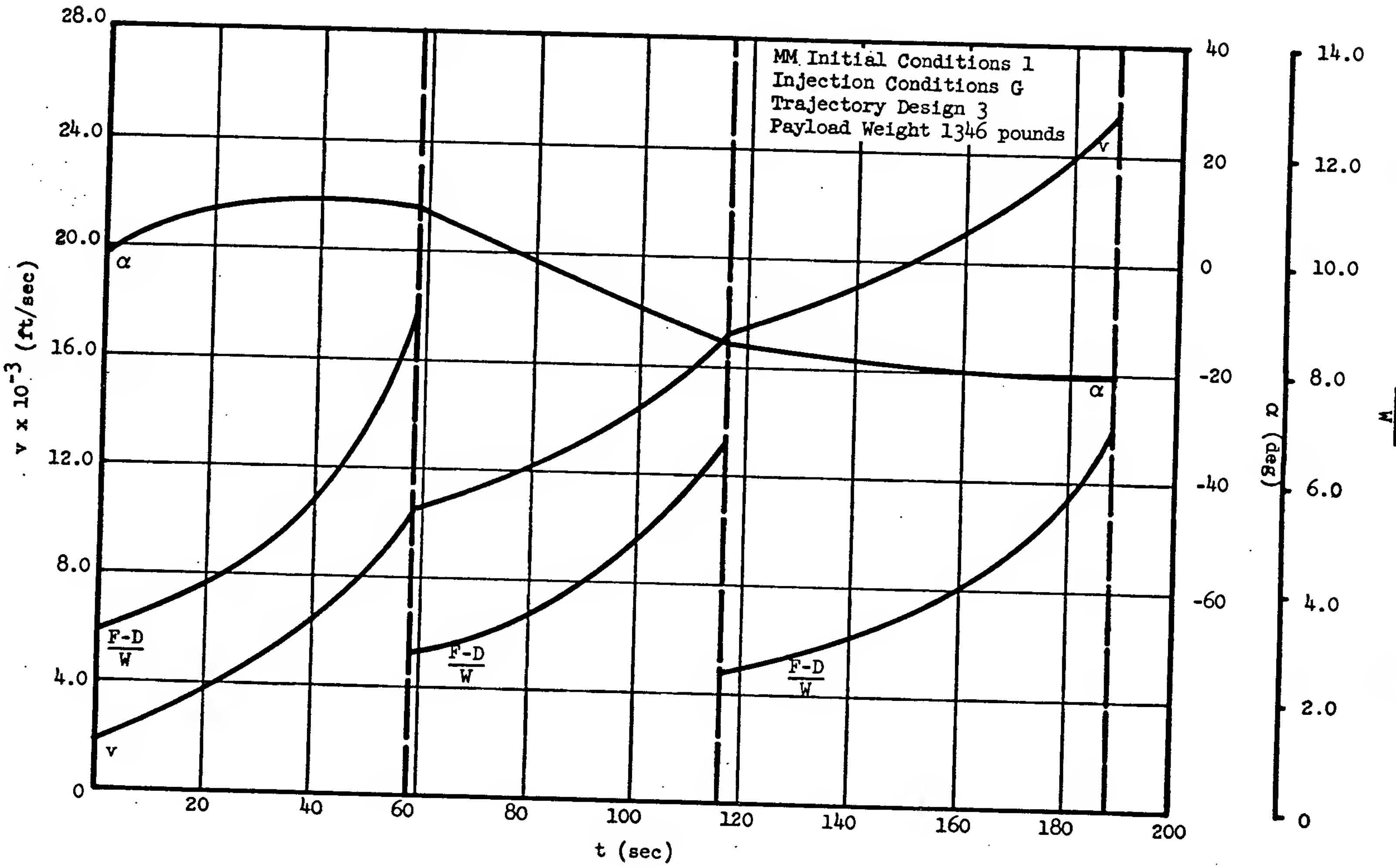

Figure G-19. Reference Trajectory for Orbital Injection Conditions H (3/4 percent ellipse)

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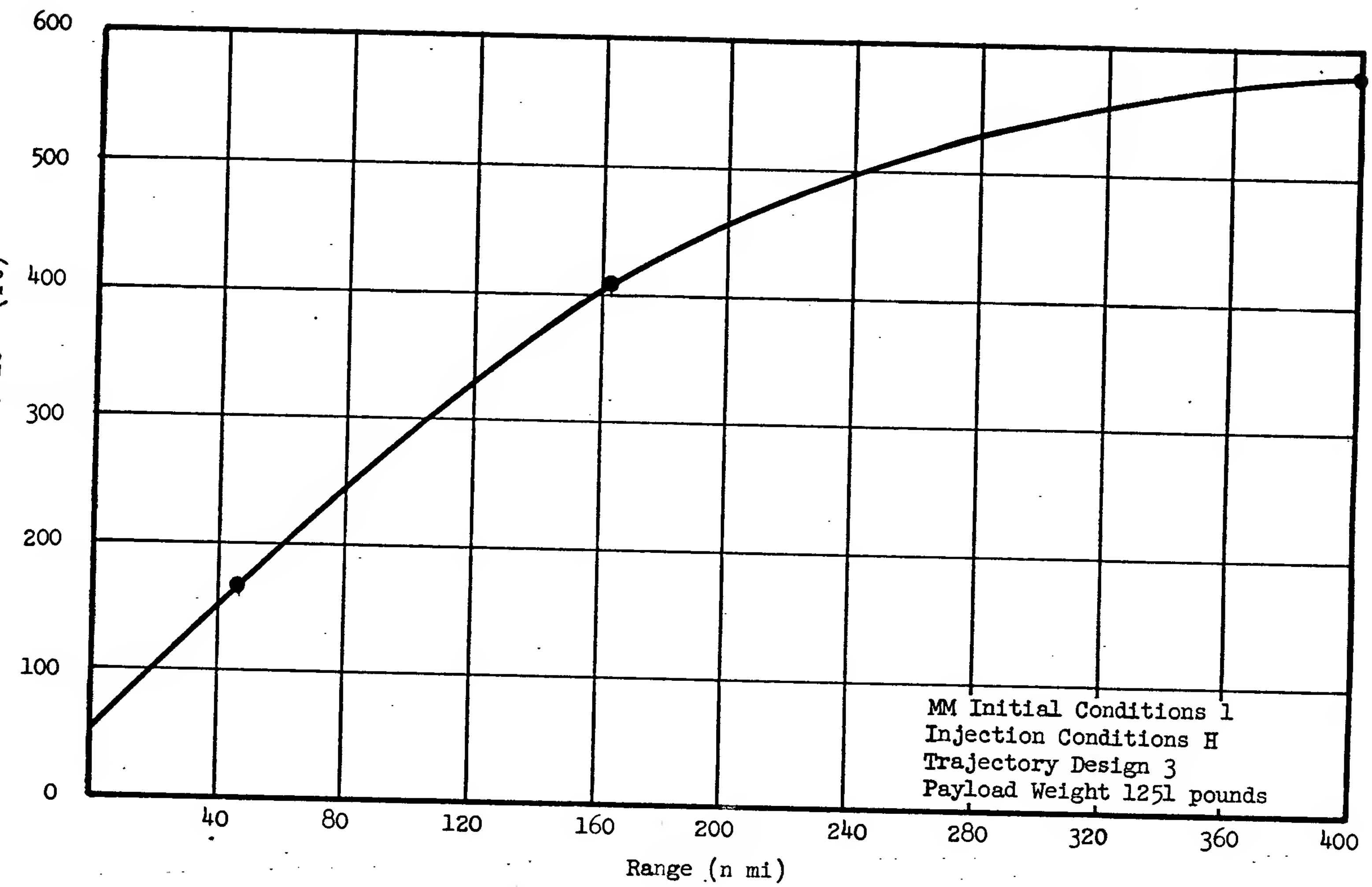

Figure G-20. Reference Trajectory for Orbital Injection Conditions H (3/4 percent ellipse)

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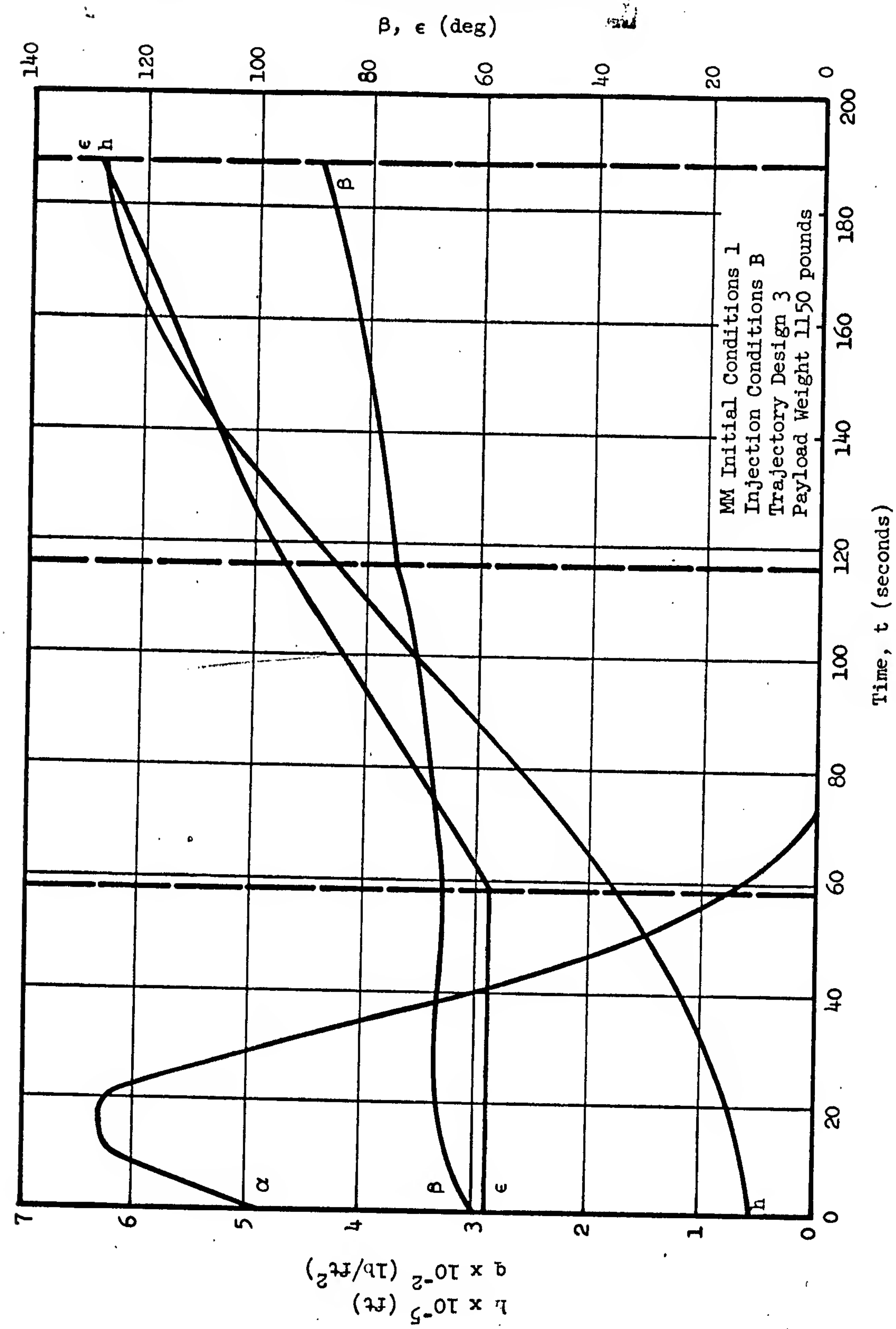

Figure G-21. Reference Trajectory for Orbital Injection Conditions B (one percent ellipse)

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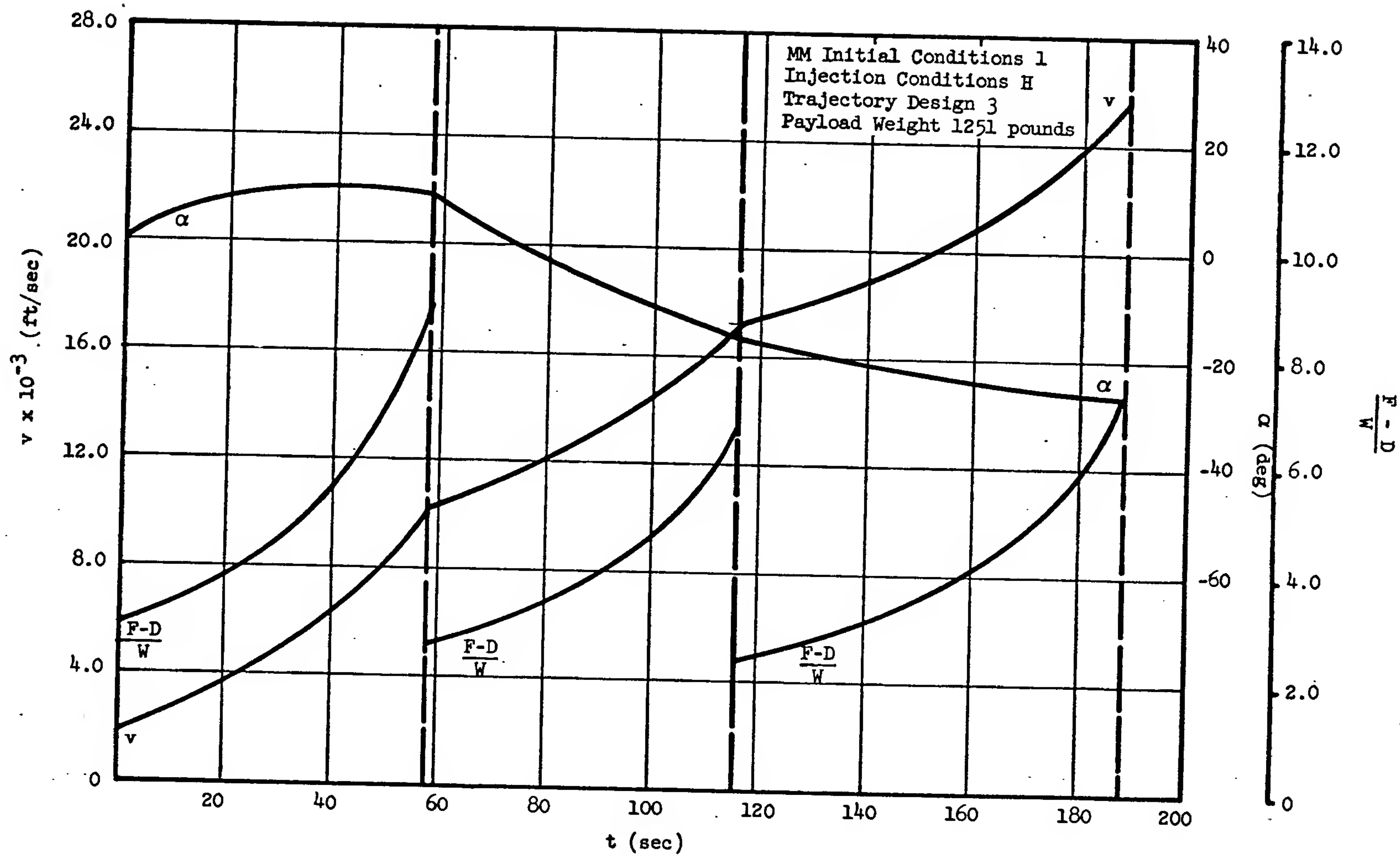

Figure G-22. Reference Trajectory for Orbital Injection Conditions B (one percent ellipse)

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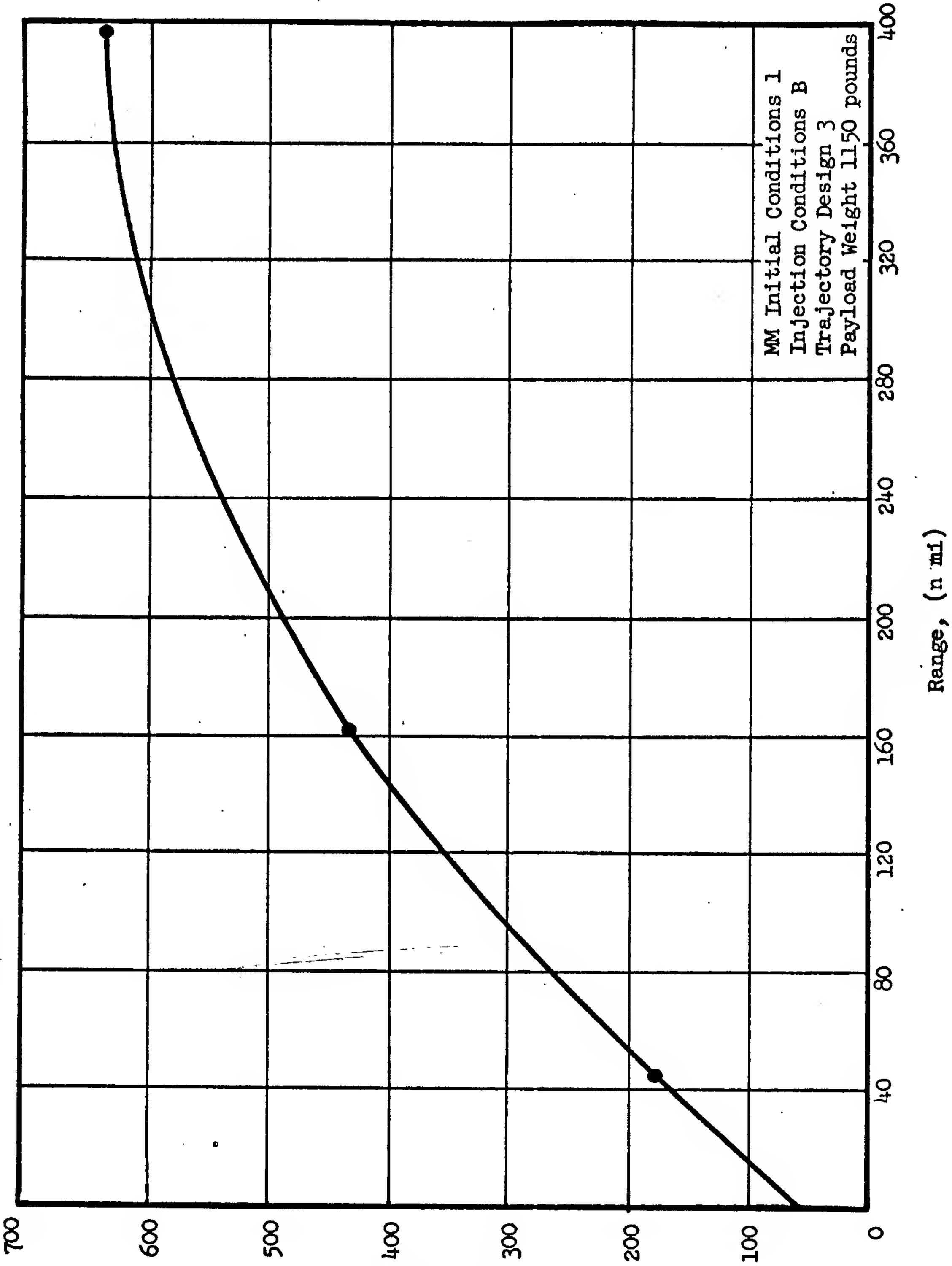

Figure G-23. Reference Trajectory for Orbital Injection Conditions B (one percent ellipse)

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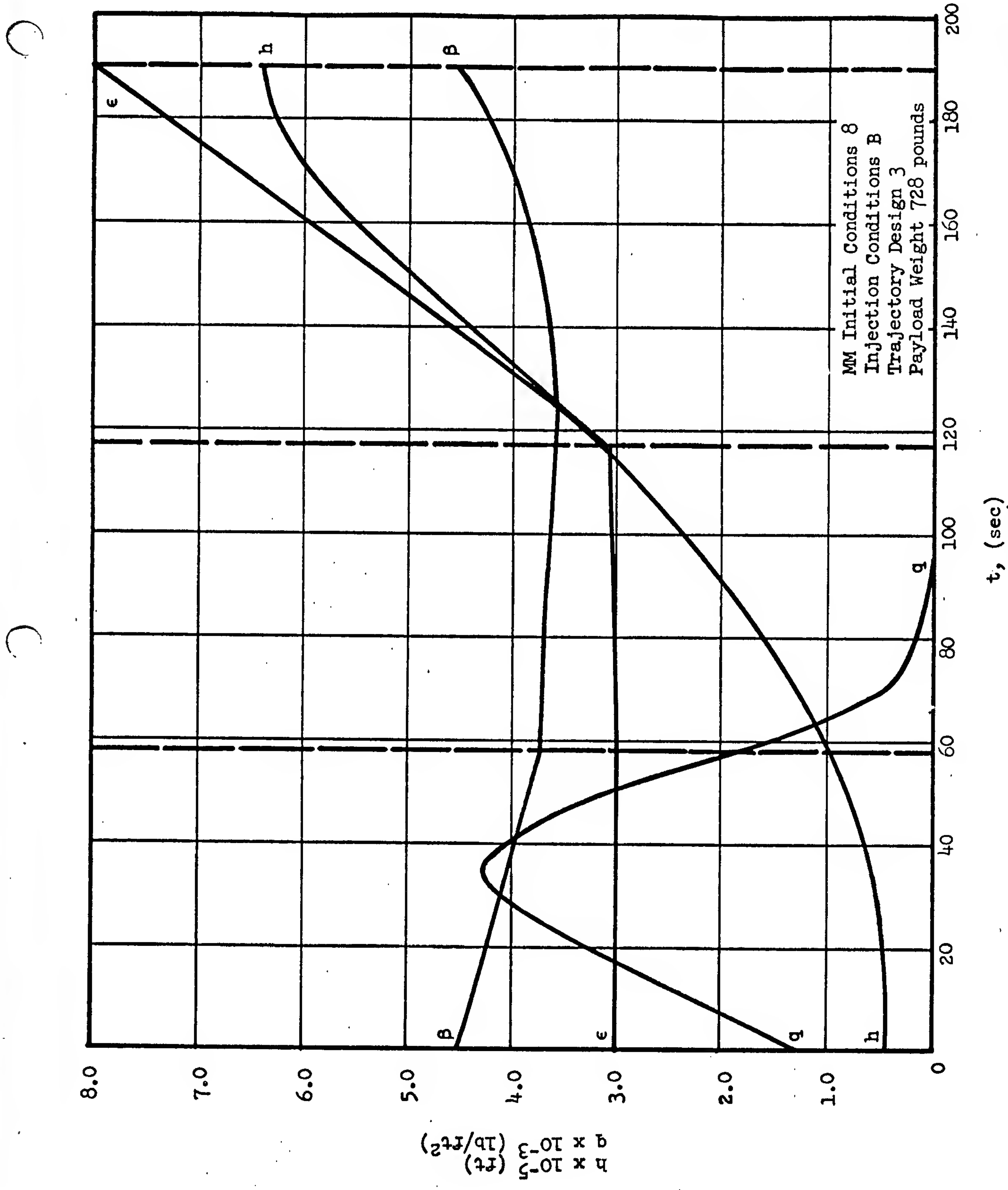

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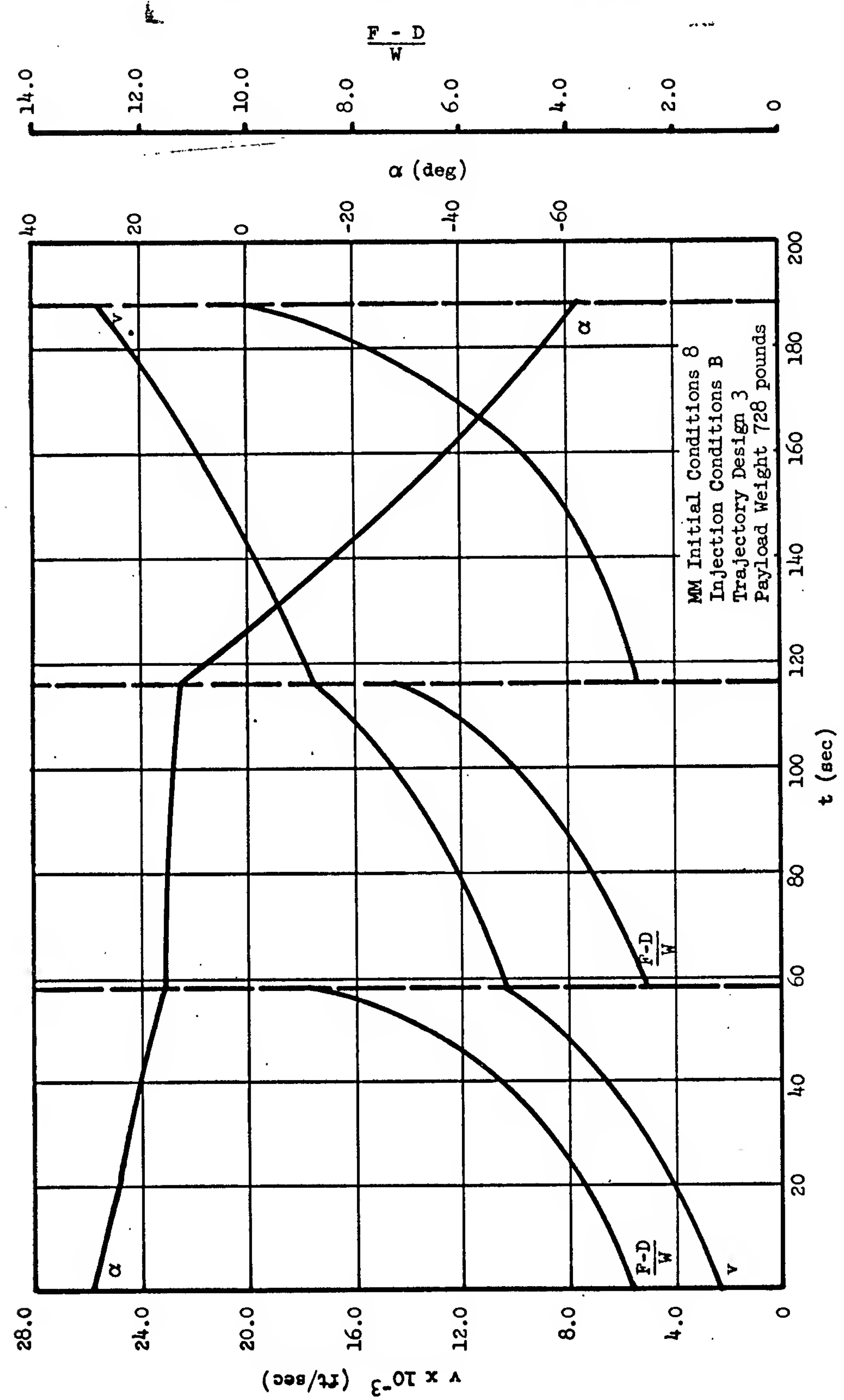

Figure G-25. Horizontal Launch Trajectory

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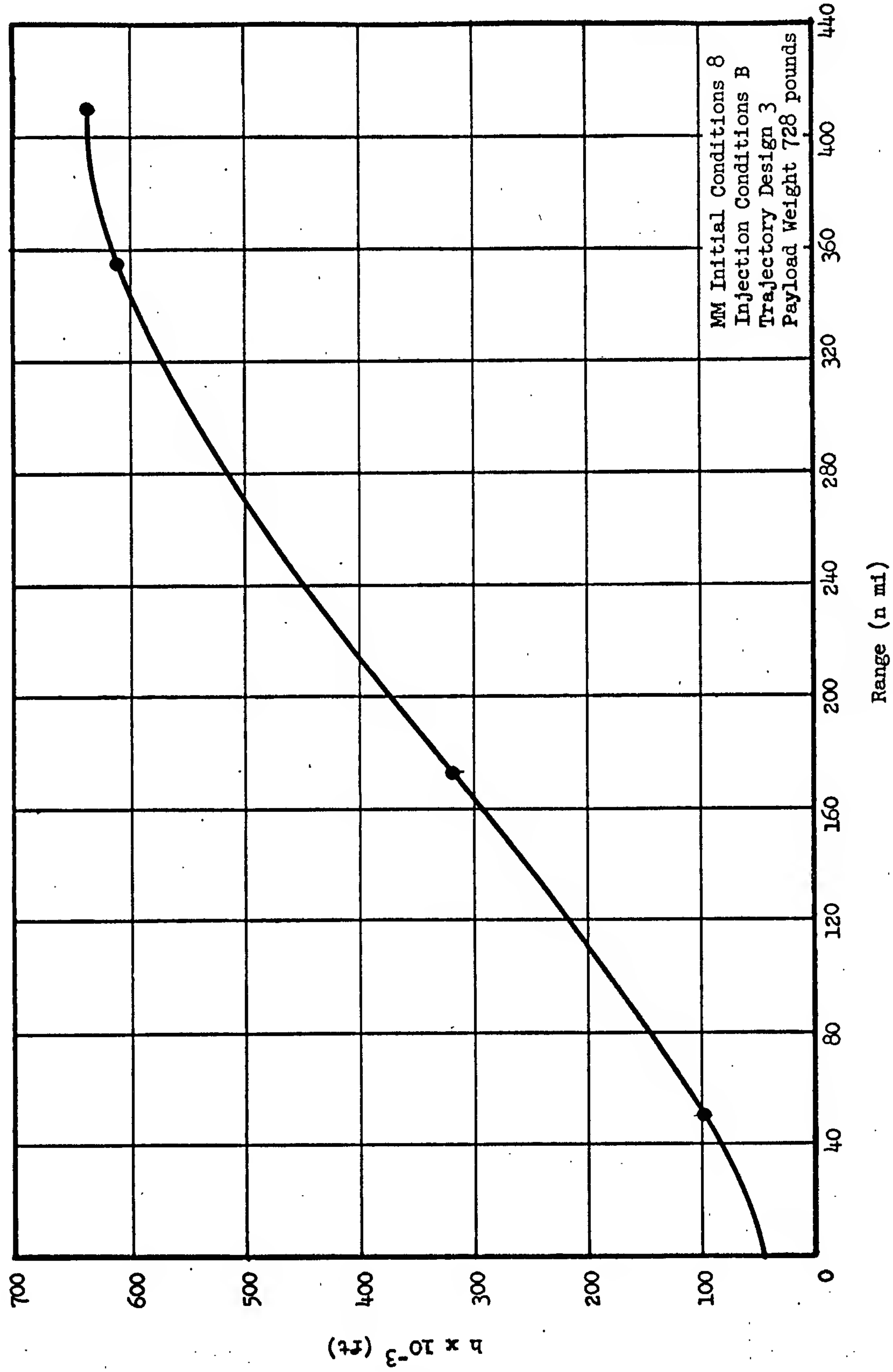

Figure G-26. Horizontal Launch Trajectory

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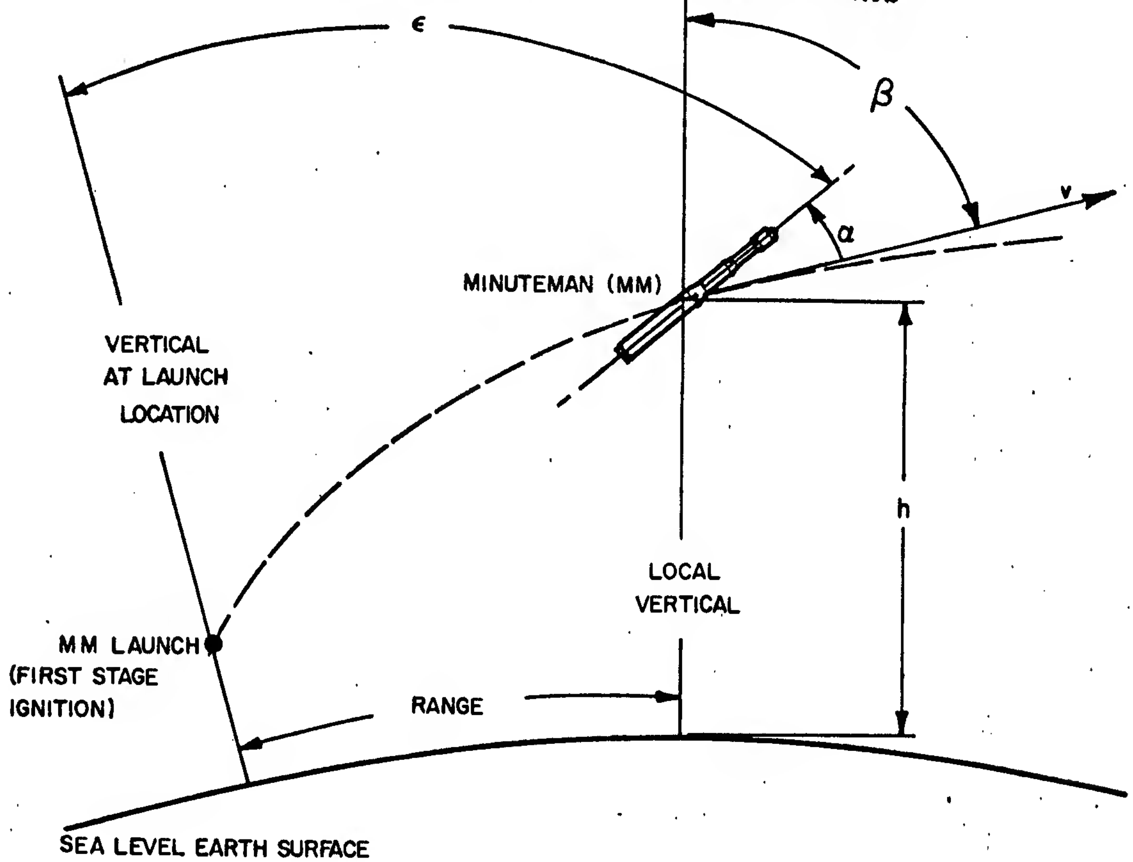

α_o = MM angle of attack at launch

α_i = Pod (and MM) angle of attack at pod release

v = MM velocity relative to nonrotating earth

v_a = MM velocity relative to atmosphere (= v)

ΔV = MM velocity increment

$\dot{\epsilon}$ = rate of change of ϵ with respect to time

g = MM dynamic pressure

Injection angle = flight path angle (β) at injection into orbit (MM third stage burnout)

Inclination angle = angle of the orbiting flight path with respect to the earth's equator on the first north-south pass (eastward along the equator = 0)

Launch = MM firing = time at MM first stage ignition

Pod release = time at separation of pod from B-58A

$$\text{Percent ellipse} = \frac{\text{apogee} - \text{perigee}}{\text{apogee} + \text{perigee}} (100)$$

Figure G-27. Definition of Terms

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APPENDIX H

LIFETIME STUDIES

It is a well-known phenomenon that due to atmospheric drag effects low altitude orbits have relatively short lifetime expectancies. The low altitude, low eccentricity orbits proposed for the operational system characteristically spiral in slowly even during the first few orbits (see Figures H-1 thru H-7). The problem here is to determine the useful lifetime, in terms of the number of orbits for the satellite, which can be obtained from various orbits selected for study. The objective is to have at least three orbits useful for photographic purposes. To be useful for photographic purposes the altitude over the target must be between the highest altitude from which the desired photographic resolution can be obtained and the minimum altitude dictated by atmospheric drag and heating considerations.

The STL tracking program was used in this study to calculate the orbital path of the satellite. This computer program solves the equations of motion as the vehicle orbits around an ellipsoidal earth with an ARDC 1959 atmosphere.

All orbital simulations for lifetime studies were for 70° inclined orbits with the satellite traveling from west to east. With a 70° inclined orbit, coverage is possible for nearly all desired target areas assuming southeasterly launches in the general ocean area between the North American Continent and the Hawaiian Islands. This launch area presupposes the use of inertial guidance for the Minuteman ascent trajectory. Using radio guidance with the ground station on a Pacific Island (such as Midway, the Hawaiian Islands, Flint, Vostock or Carolina Island), the orbital inclination is determined by the target areas desired. For lower orbital inclination, the lifetime calculations are conservative, assuming the same

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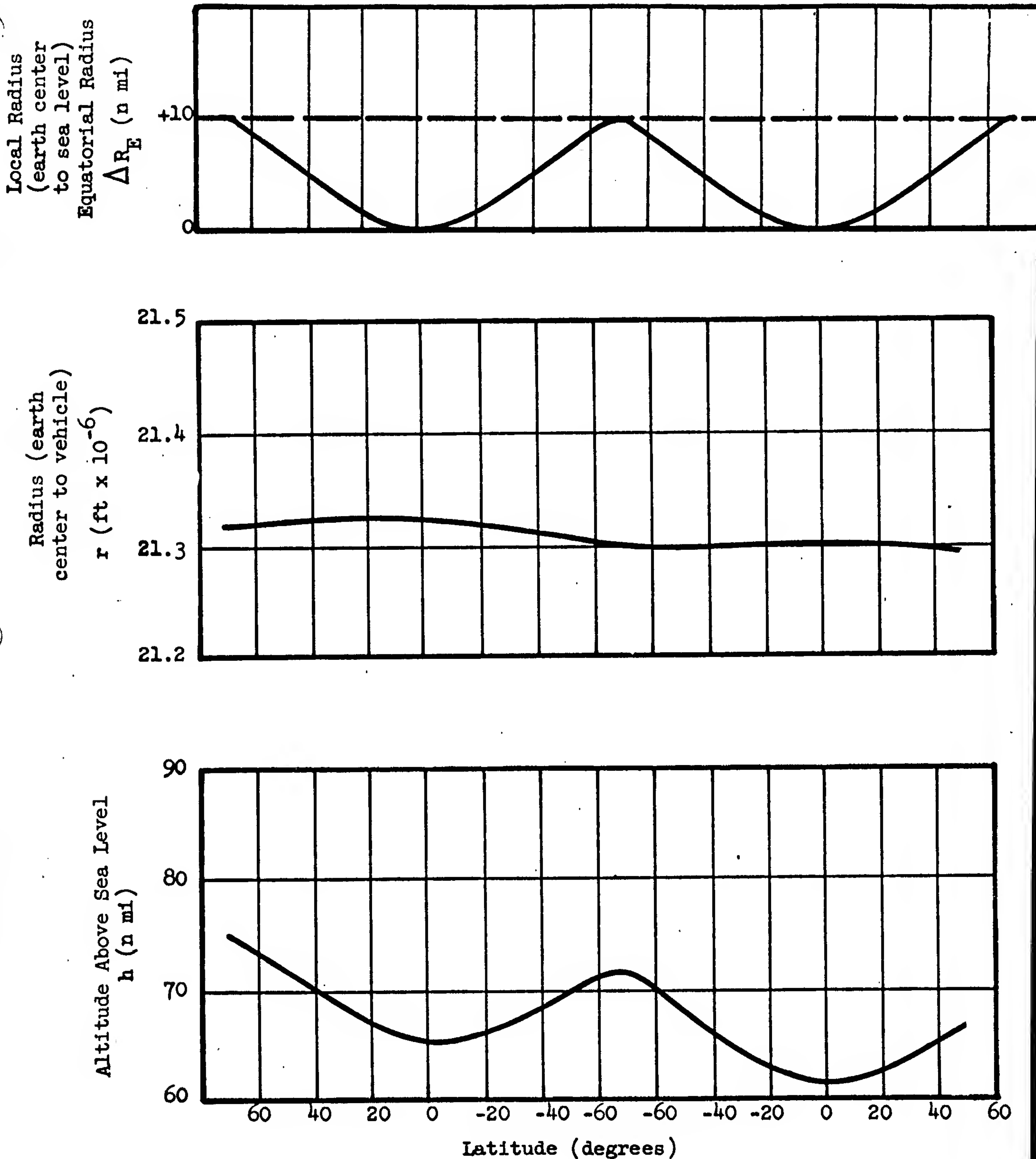

Figure H-1. Lifetime Orbital Altitudes, 75 Nautical Mile Circular Orbit, $W/C_DA = 50$ (lb/ft^2) (simulation no. 21)

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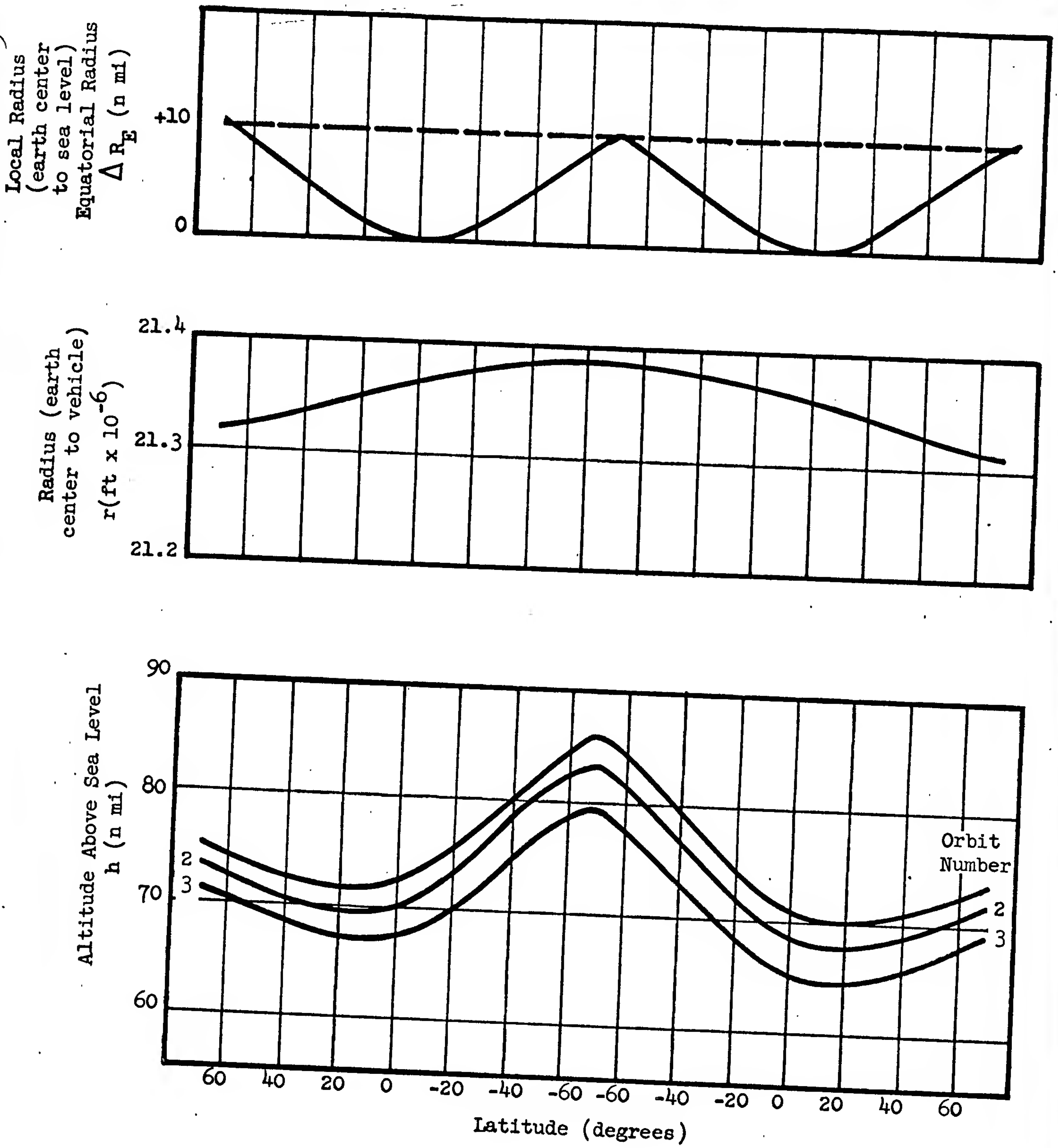

Figure H-2. Lifetime Orbital Altitudes, 75 Nautical Mile Perigee, Elliptical Orbit (ϵ' = 0.002), W/C_DA = 50 lb/ft^2 (simulation no. 22)

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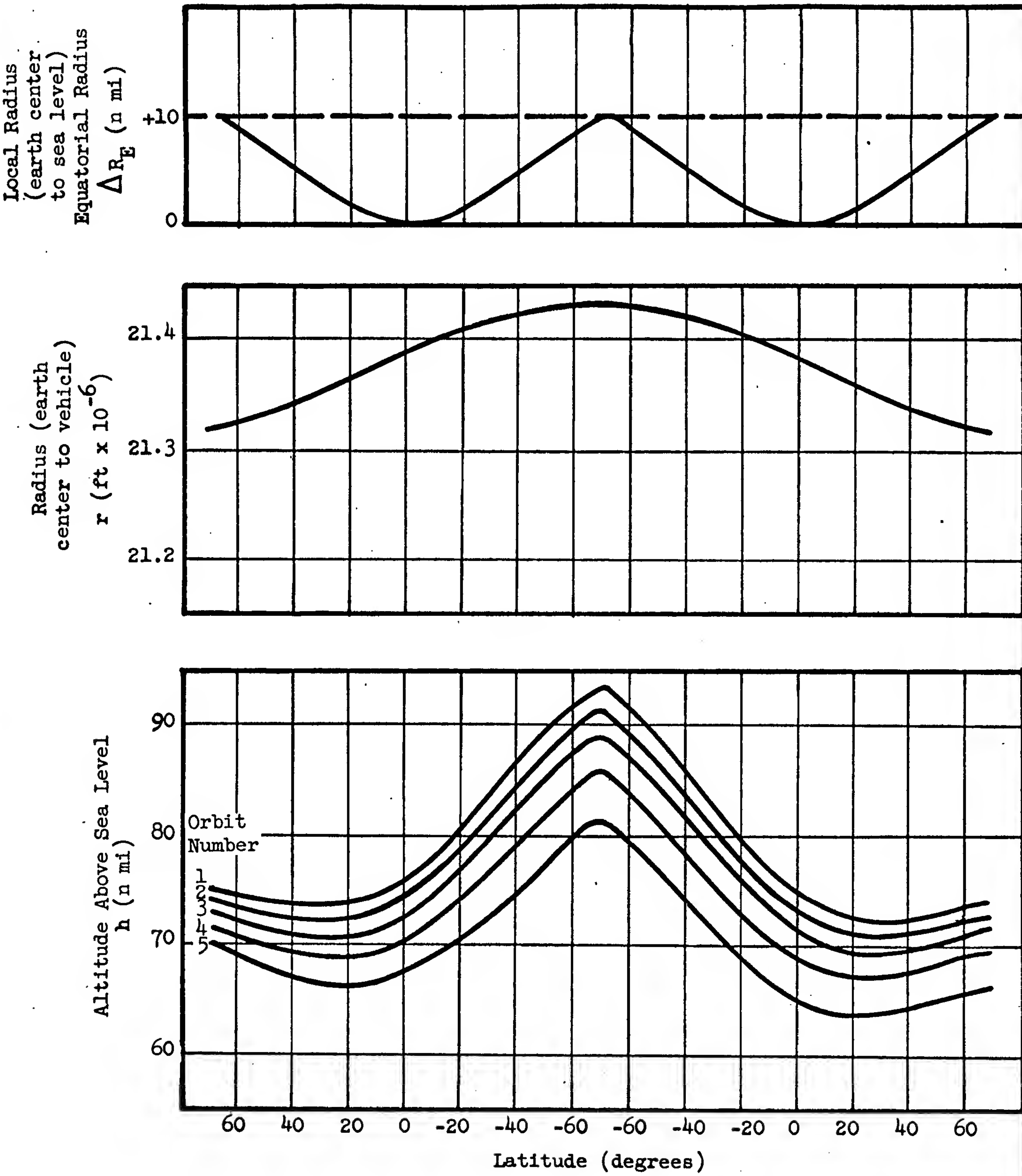

Figure H-3. Lifetime Orbital Altitudes, 75 Nautical Mile Perigee, Elliptical Orbit ($\epsilon' = 0.003$), $W/C_DA = 50 \text{ lb/ft}^2$ (simulation no. 23)

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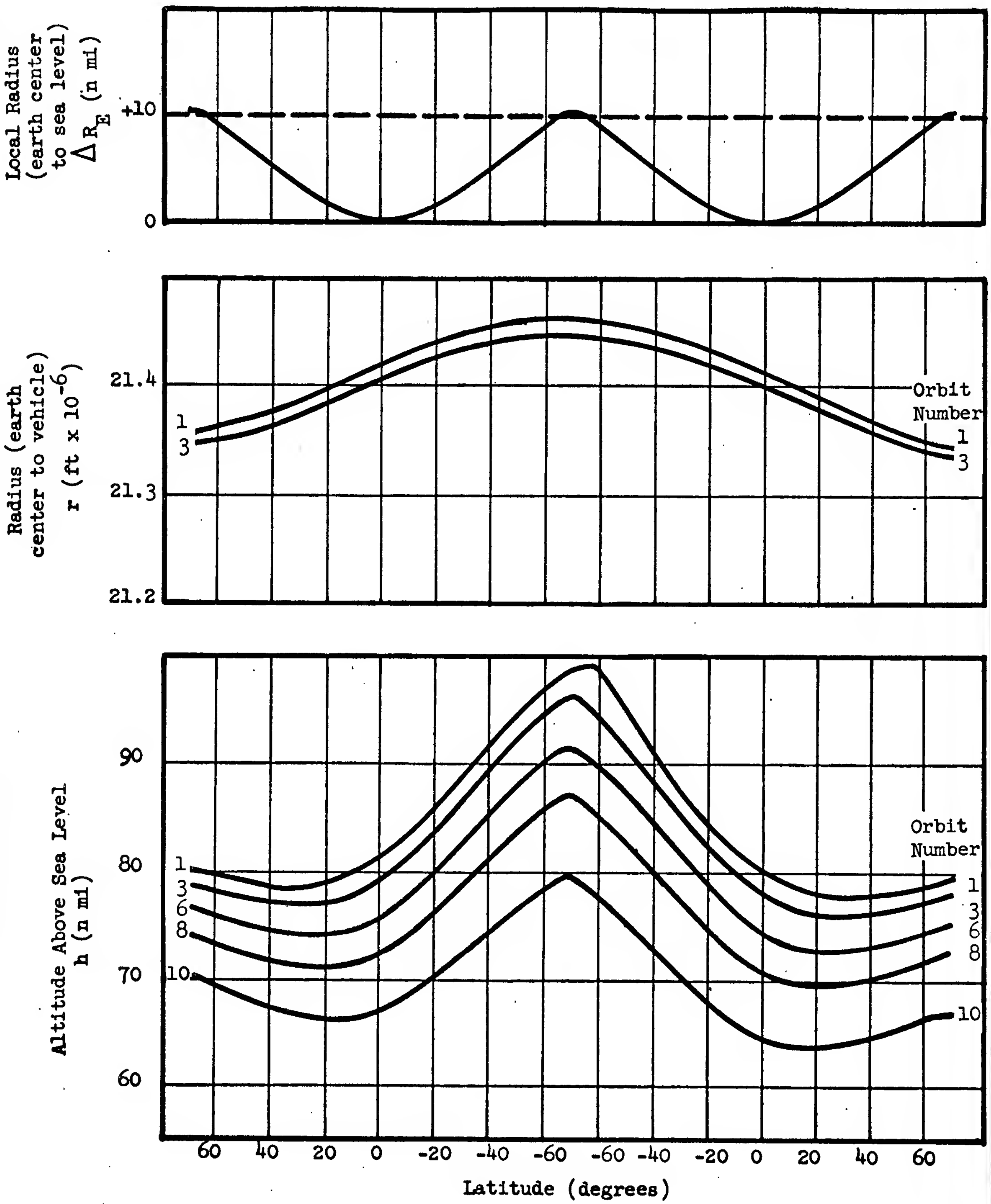

Figure H-4. Lifetime Orbital Altitudes, 80 Nautical Mile Perigee, Elliptical Orbit ($\epsilon' = 0.003$), $W/C_D A = 50$ lb/ft^2 (simulation no. 33)

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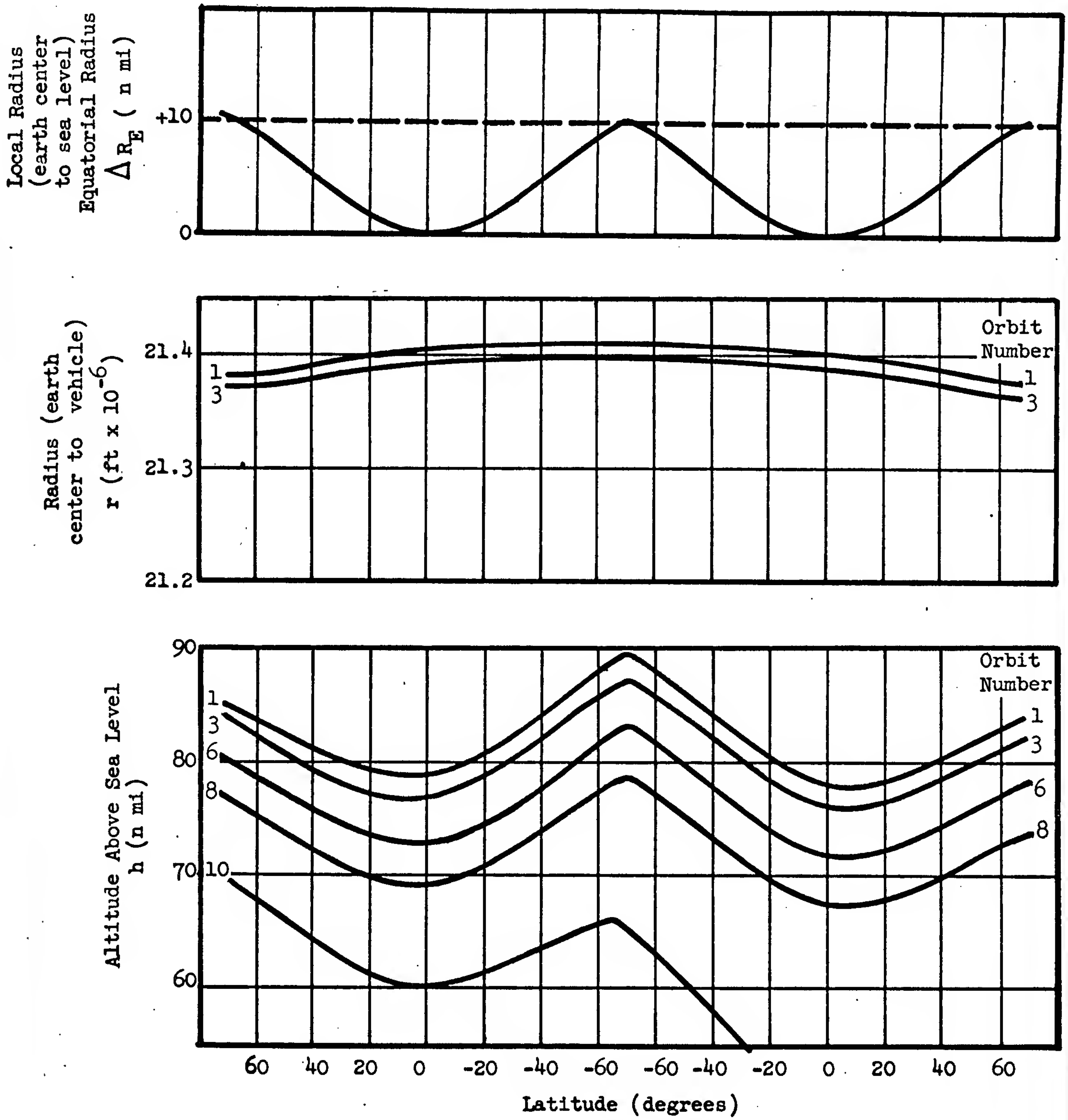

Figure H-5. Lifetime Orbital Altitudes, 85 Nautical Mile Perigee, Elliptical Orbit ($\epsilon' = 0.001$), $W/C_DA = 50$ lb/ft^2, (simulation no. 35)

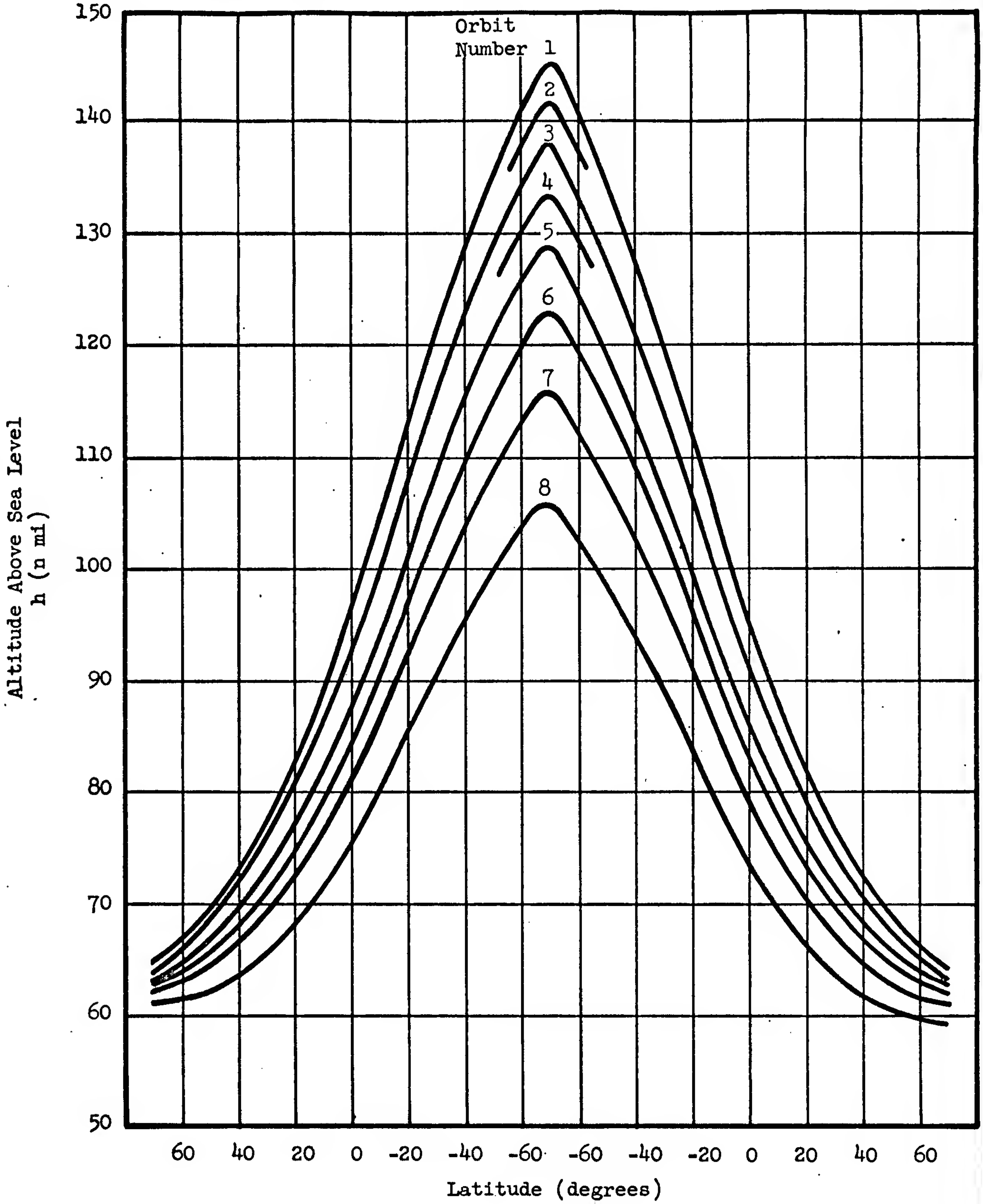

Figure H-6. Lifetime Orbital Altitudes, 65 Nautical Mile Perigee, Elliptical Orbit ($\epsilon' = 0.012$), $W/C_DA = 50$ lb/ft^2, (simulation no. 28)

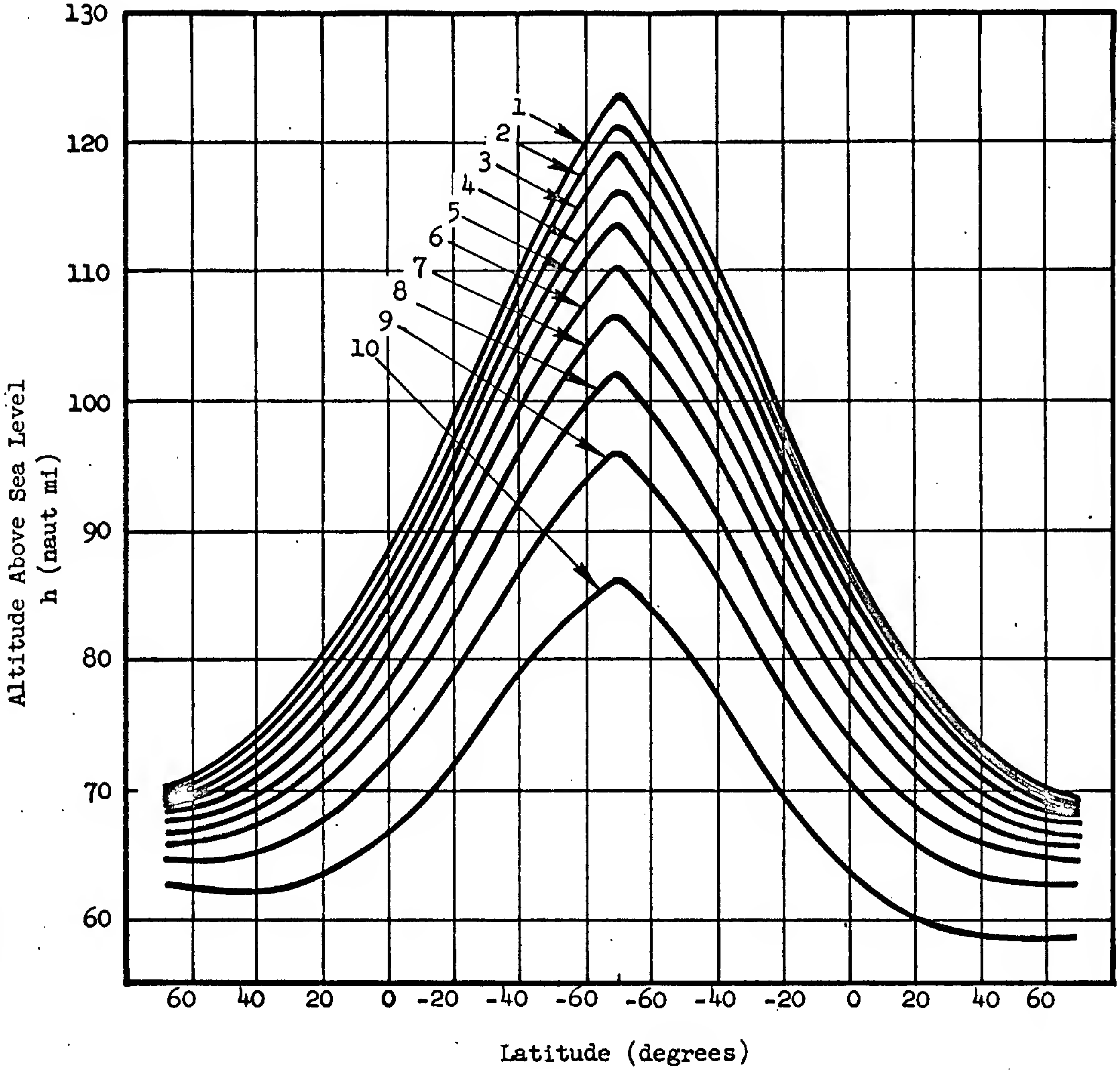

Figure H-7. Lifetime Orbital Altitudes, 70 Nautical Mile Perigee, Elliptical Orbit ($\epsilon' = 0.008$), $W/C_DA = 50$ lb/ft^2, (simulation no. 31)

minimum orbital altitude and orbit eccentricity are maintained. The reason for this is found in the effects of the ellipsoidal shape of the earth on the height of the orbit above sea level (see Figures H-1 through H-7). The curve at the top of these figures represents the equatorial radius of the earth minus the local radius of the earth (ΔR_E). The curve of the center of the page represents the orbital radius from the center of the earth (r). The bottom curves represent the height of the orbiting vehicle above sea level on the ellipsoidal earth (h). The numbers labeling the curves and apogee and perigee points represent the first, second, third, etc., orbit around the earth. The pronounced effect of ΔR_E on h for these 70° inclined orbits is very evident. For a desired minimum altitude over the target area (say 70 nautical miles), the path of the orbiting vehicle is within 2.5 nautical miles of the desired altitude during its entire passage through the northern latitudes (see Figure H-3).

The maximum value of the quantity ΔR_E decreases for orbits of lower inclinations under the conditions described above. This then decreases the effect of ΔR_E which keeps h low over a high percentage of its path. In the limit, an orbit of 0° inclination (equatorial orbit) would have $\Delta R_E = 0$ and the altitude (h) curve would have the same shape as the orbital radius (r) curve. These lower inclination orbits would experience lower aerodynamic drag effects (aerodynamic drag is roughly proportional to air density which is decreasing by an order of magnitude with each 5 nautical miles increase in altitude) and would therefore have a longer lifetime expectancy.

The orbital direction of west to east is used because of the desirability of launching to take advantage of the earth's rotational velocity. East to west orbits would decrease orbital lifetimes slightly because of the effect of the

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rotating atmosphere.

For lifetime study purposes, all orbits were started at perigee in order to simplify the determination of the orbital initial conditions. This makes the lifetime results slightly conservative with respect to the orbits for the operational system which will start in the neighborhood of 90° to 110° after perigee. Perigee is located over the target areas.

Perigee is defined as the point in the orbit where the orbital radius (r) is a minimum. This does not necessarily correspond to the point(s) in the orbit where altitude above sea level on the ellipsoidal earth is a minimum.

The eccentricity of these orbits is defined as the theoretical eccentricity of the orbit calculated for a spherical earth and zero aerodynamic drag.

$$\epsilon' = \frac{r_a - r_p}{r_a + r_p}$$

ϵ' = orbit eccentricity

r_a = apogee radius

r_p = perigee radius

The actual orbits are osculating in nature and difficult to describe in exact terms. The orbital simulations and lifetime results are presented in Table H-1. The expected maximum lifetimes to impact are plotted in Figures H-8 and H-9.

These results show that lifetime is exactly proportional to W/C_DA with the exception of very short lifetimes (less than 3 orbits). For example:

Simulation Number	Perigee Altitude (n mi)	Eccentricity	W/C_DA (lb/ft^2)	Lifetime to Impact (Minutes)	(Orbits)
22	75	0.002	50	362	4.16
37	75	0.002	30	215	2.47
18	65	0.005	50	200	2.3
19	65	0.005	30	137	1.6

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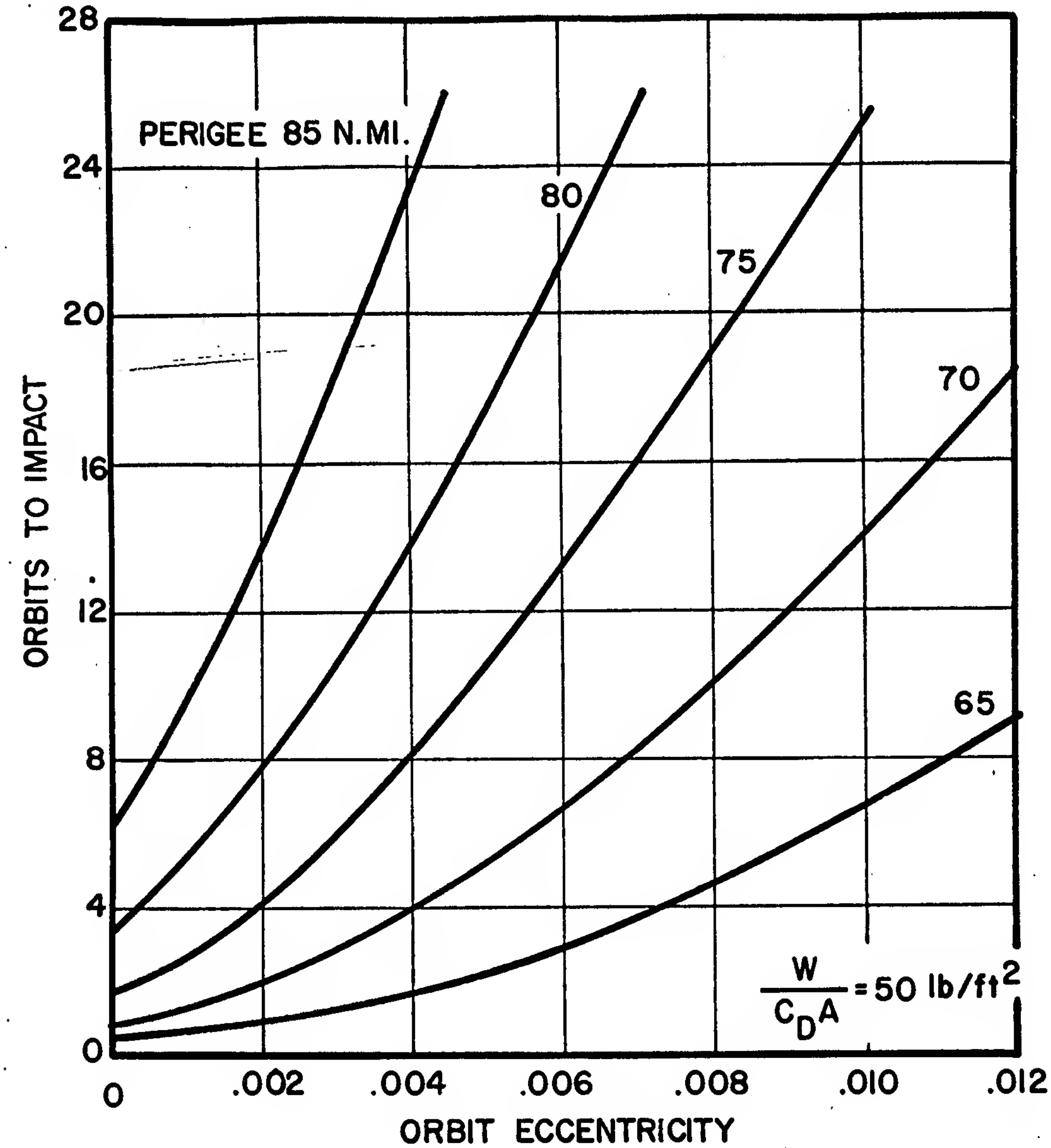

Figure H-8. Low Altitude Orbital Lifetime, Eccentricity = 0 to 0.012

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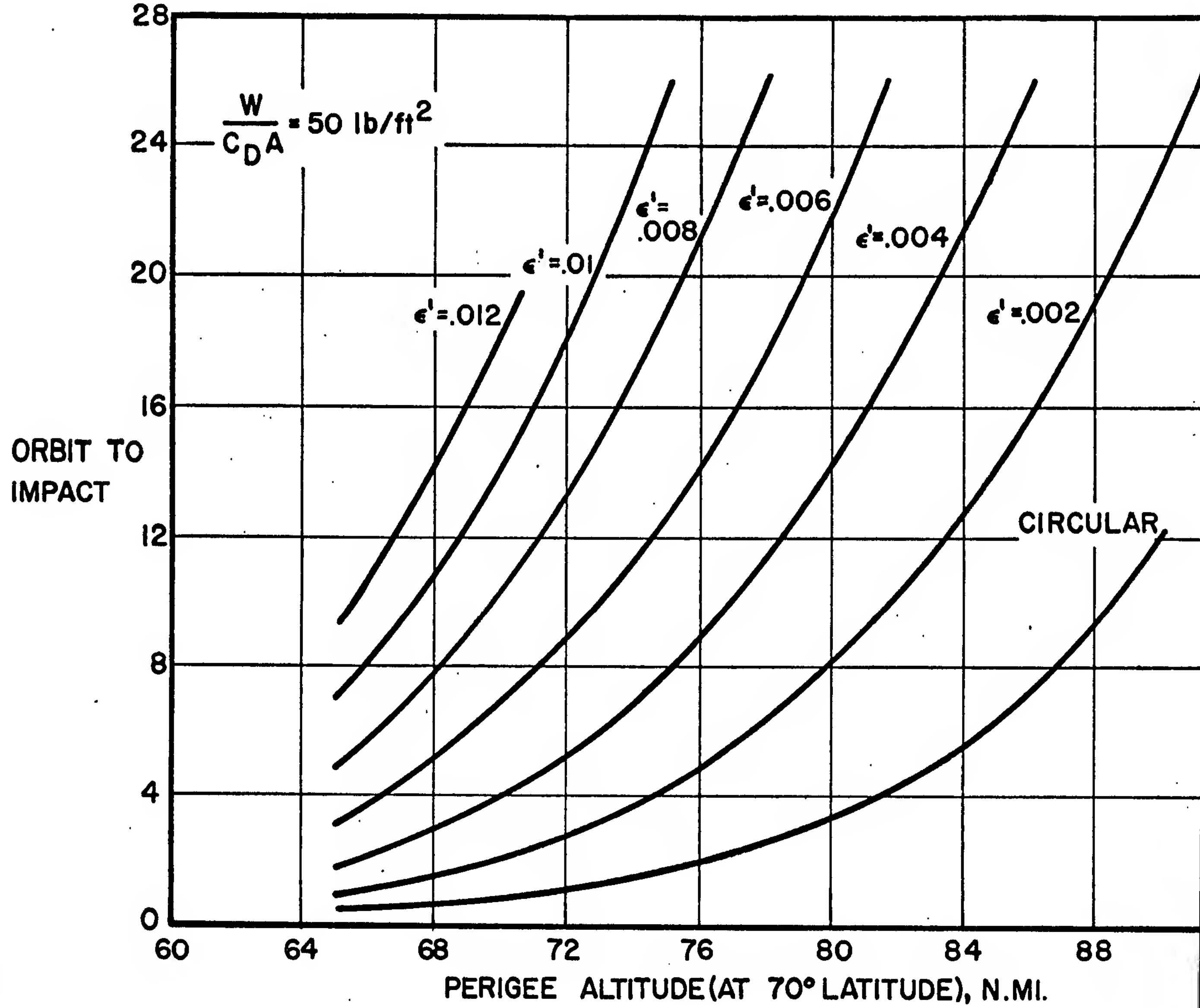

Figure H-9. Near Circular Orbital Lifetime, Perigee = 65 to 90 Nautical Miles

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If the lifetime W/C_DA were constant, the lifetime to impact for the $W/C_DA = 30$ simulations would have been 217 minutes for simulation number 37 which agrees with the calculated value (215 minutes) within the accuracy of these calculations, and 120 minutes for simulation number 19 which is significantly lower than the calculated value (137 minutes).

Figures H-2 through H-7 have been included to show the orbital altitude above sea level as it is effected by the ellipsoidal earth, and the loss of energy by the orbiting vehicle from orbit to orbit as it progresses through its orbital life, for several representative orbit simulations. The simulation which most nearly represents the desired orbit for the operational system is simulation number 33 (first perigee altitude see Figure H-4, 80 nautical miles, eccentricity = 0.003). Orbit altitudes over target areas (latitude 20° N to 70° N) for the first three orbits varies from 75.7 to 80 nautical miles with $W/C_DA = 50$ lb/ft^2. The satellite W/C_DA is estimated to be 40 lb/ft^2 which it is estimated will bring down the minimum altitude to 75.3 nautical miles. The maximum nominal altitude desired for photographic purposes is 75 nautical miles (450,000 feet). Lifetime to impact is 8.6 orbits with $W/C_DA = 40$ lb/ft^2. The orbital injection conditions (typical for injection in the neighborhood of 20° N to 30° N latitude) are:

Injection latitude	26.3°
Injection altitude	78.2 nautical miles
Injection velocity	25,673 ft/sec
Flight path angle at injection (β)	89.84°

For these injection conditions, the B-58A (maximum level flight Mach number = 2.0)/Minuteman (Wing II degraded 3 sigma) combination can orbit 1400 pounds with a southeasterly launch using a continuous burn trajectory. This 1400 pounds includes guidance system weight, adapter section and satellite.

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Table H-1. Orbit Simulation for Lifetime Study

Simulation Number	Orbit Description Perigee (n.Mi.)	Eccentricity (ϵ')	Injection Velocity* (ft/sec)	W/C_DA (lb/ft^2)	Number of Orbits to Impact	Notes
1	69.77	0.0014	25,709	50	1.9	(1)
2	69.77	0.0064	25,773	50	8.2	(1)
3	69.77	0.0112	25,835	50	18.5	(1)
4	69.77	0.0014	25,709	100	3.8	(1)
5	69.77	0.0064	25,773	100	16.5	(1)
6	69.77	0.0112	25,835	100	37.0	(1)
7	79.77	0.0013	25,672	50	7.5	(1)
8	79.77	0.0049	25,718	50	20.2	(1)
9	79.77	0.0099	25,781	50	45.5	(1)
10	79.77	0.0013	25,672	100	15.2	(1)
11	79.77	0.0049	25,718	100	-	(1)(2)
12	79.77	0.0099	25,781	100	-	(1)(2)
13	89.77	0.0014	25,636	50	22.0	(1)
14	89.77	0.0035	25,663	50	37.0	(1)
15	89.77	0.0085	25,726	50	-	(1)(2)
16	65.02	0.0025	25,740	50	1.1	(3)
17	65.02	0	25,708	50	0.5	(3)
18	65.02	0.005	25,772	50	2.3	(3)
19	65.02	0.005	25,772	30	1.6	(3)
20	65.02	0.0075	25,804	50	4.3	(3)
21	75.02	0	25,672	50	1.63	(3)
22	75.02	0.002	25,692	50	4.16	(3)
23	75.02	0.003	25,710	50	6.03	(3)
24	77.79	-0.0005	25,655	50	1.82	(3)
25	77.79	0.0015	25,680	50	4.9	(3)
26	77.79	0.0025	25,693	50	7.2	(3)
27	89.85	0	25,618	50	12.1	(3)
28	64.85	0.012	25,861	50	9.1	(3)
29	69.85	0	25,691	50	.8	(3)
30	69.85	0.004	25,742	50	4.0	(3)
31	69.85	0.008	25,793	50	10.2	(3)
32	79.85	0	25,654	50	3.3	(3)
33	79.85	0.003	25,692	50	10.8	(3)
34	84.85	0	25,635	50	6.3	(3)
35	84.85	0.001	25,648	50	9.8	(3)
36	84.85	0.003	25,674	50	18.7	(3)
37	74.85	0.002	25,697	30	2.47	(3)

* at first perigee
(1) Perigee located at 60° N Latitude (ahead of 70° N max. latitude)
(2) Simulation stopped before impact
(3) Perigee located at 70°N latitude

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APPENDIX I

CAMERA-OPTICAL SYSTEM

This appendix is concerned with the feasibility of designing a camera-optical system for a reconnaissance satellite that is capable of obtaining high quality photographic information on a specific target within the limitations of payload volume and weight associated with the air launch mission under consideration.

It was possible to use the results of a parallel study effort on a ground-launched reconnaissance satellite employing an 80-inch optical system as the basis for considering the feasibility of the camera-optical system for the air-launched mission. Since the orbital payload capability of the air-launched system is less than that attainable with the ground launched system considered for use with the 80-inch system, the optical-camera system must be reduced in size and weight. This produces some reduction in the photographic system performance. The following discussion evaluates only the feasibility of this system concept and does not attempt to establish fully detail design parameters.

The photographic instrument package for the air-launched satellite is based on the use of two cameras, oriented $\pm 20^{\circ}$ from vertical for stereo photography. These two cameras operate in a strip camera mode, continuously photographing a 10 nautical mile wide strip over the area of interest, with active correction for the image movements which would otherwise degrade the results. It is expected that this camera package will provide average effective ground resolutions of the order of 3 feet.

The optical design of the cameras will be based on the use of objectives with a 40-inch equivalent focal length and with a T-stop of 5.0 or better. The objectives currently contemplated for use are of the off-axis catadioptric type,

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with diffraction limited performance. Advantage will be taken of the current design program for an 80-inch F/5 off-axis catadioptric system. Scaling down will afford some design improvements. Design goals are aimed at achieving a low contrast lens-film resolution performance of 200 lines/mm.

Other objectives will be considered, such as an on-axis catoptric (all mirror) system, and also a Petzval type refractor system. Current investigations, however, tend to preclude their use. In the satellite, the two cameras are mounted as a unit with their optical axes inclined to each other at an angle of 40° to provide stereo photography. One camera looks forward along the flight path and the other rearward (see Figure J-2 in Appendix J).

The use of a fully stabilized satellite makes practical the use of strip cameras which, due to their simplicity and low film transport velocity, have the best reliability. Due to the attitude stabilization, recovery considerations, lifetime, and shape of the satellite, the cameras are folded so the satellite can travel apex forward. For the focal lengths planned, the use of the additional mirror in each camera to fold the optical path will not result in appreciable loss of basic resolution.

Each camera is fitted with a minus blue filter and also will be fitted with a polarizing screen if current investigations prove that higher emulsion speeds are obtainable by direct positive transparency processing and that there is no appreciable inherent optical resolution loss by use of the screen. The purpose of the polarizing screen is to improve image contrast by removal of that scattered light in the atmosphere which, for most orbit inclinations, is strongly polarized.

The major portion of each optical system operates under accurate temperature control (as described elsewhere). A small window is fitted near the focal plane

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so that the film transport can take place in an atmosphere of 14.7 ± 0.5 psia and a relative humidity of 17 to 25 percent at 70^{o}F. One purpose of this environment is to provide maximum film transport reliability.

The camera structure is based on a closed end box, formed of sheet Invar, for maximum stiffness and temperature stability. Each camera tube or box is fitted with a number of baffles to reduce veiling glare to the absolute minimum.

The film transport system for each camera consists of film supply and take-up cassettes and a digital film drive which is dynamically controlled to provide for compensation of image degrading movements, which are above the optical noise level.

Present thinking contemplates the design of individual film cassettes to provide heat protection during re-entry and also protection from exposure to light. Further, the use of light-tight cassettes very materially facilitates testing and final loading of the cameras. In addition to protecting the film, the cassettes are also fitted with metering rollers for film supply and take-up. Further weight reduction may be afforded by making the recovery capsule an integral film cassette which provides full mechanical and light protection for the film of both cameras.

The choice of film and film processing will be determined by current investigations. The desirable choices center around the use of Kodak SO 243 type emulsion on a thin polyester base with a thickness of 2.5 mil. Processing will be of the direct positive transparency type if possible. There are two main reasons for this--to correct for exposure errors during process and thus retain maximum resolution, and to permit interpretation of the original film and thus obviate the loss of some of the information content which is generally concomitant

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with any negative-to-positive reporduction process. In any event, it is contemplated that the interpreter will be supplied with a positive transparency for optimized interpretability.

The film transport for each camera is mechanically designed to keep the film in one plane with no corners, and has special film edge guidance provisions to reduce the possibility of edge scalloping or tearing to an absolute minimum. Rollers are of adequate diameter so as to hold pressure marks to a minimum. Analogous provisions are also made in the design of the film gate or exposure area. Recording of vehicle attitude, time, film shrinkage, film velocity, etc., is accomplished on the edges of the film as it travels through the camera.

Detail techniques of film transport, image motion compensation, allowance for keystone effects, lens mounts, adjustments, etc., are being developed in another concurrent part of this total program. The weight and size estimates, power requirements, thermal requirements, etc., for the cameras of this design are derived from these other efforts.

To develop some of the major optical system parameters such as the focal length and f/stop for the new system configuration, several satellite configuration studies were undertaken and completed. However, consideration was made of only one specific optical system configuration. The configuration adopted for this study is a scaled-down version of the 80-inch E.F.L.-f/5.0 catadioptric system, where the effective focal length has been reduced to 40 inches and f/5.0 has been retained. The best satellite configuration contains two photographic instruments mutually oriented for stereo-photography.

It should be pointed out, however, that a 40-inch E.F.L.-f/5.0 catadioptric optical system configuration has been used in all of the present studies only for reasons of expediency. It is important to realize that a change in a focal length

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requirement from 80.0 inches down to 40.0 inches constitutes a major modification in the scope of optical design. To achieve the best system of 40.0 inch focal length from the standpoint of performance, size, and weight, other optical system configurations must be re-evaluated. This applies particularly to the refractor, since all weights have theoretically now been reduced by a factor of 8 plus the fact that a refractive type optical system has a physical length dimension nearly always equal to its effective focal length. Since the catadioptric system is generally twice as long as its effective focal length it might be possible to use a refractor in its place with a considerably larger focal length.

As a consistent design guide, we have derived for the 40-inch E.F.L.-f/5.0 optical diffraction limited system the following major parameters:

Focal length	$f = 40.0$ inches
T-number	T-5
Angular resolution	$\theta = 3 \times 10^{-3}$ milliradians
Corresponding ground resolution	1.4 feet
Corresponding blur circle diameter	3.13 μ (microns)
Corresponding resolution in image plane	320 lines/mm
On film	200 lines/mm
Net ground resolution on film	2.26 feet
Resolution to be achieved at a contrast ratio of	2:1
Assumed wavelength	$\lambda = 0.5$ microns
Field of view	$8^{\circ}42$ min
Physical configuration	Maximum 20 inch dia x 80 inches
Weight (target value)	75.0 pounds

Computations of the transfer functions for such a system have not been made in this study so it is not possible to make a more extensive evaluation at this time.

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